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56p
N63-11507
Code 1

NASA SP-17

SESSION K

Control, Guidance, and Navigation of Spacecraft

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N63-11170

27. Adaptive Guidance

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SUMMARY

The concept of the Path-Adaptive Guidance Mode is described, and illustrated with a simple example. Motivations for this approach to guidance are shown as arising out of the requirements of the Saturn space vehicle and its wide variety of missions. The way in which certain scientific disciplines are involved in the development and application of this guidance mode is discussed.

INTRODUCTION

The problem of defining a guidance system may be broken down into three areas. The first of these is the generation of information about the instantaneous state of the vehicle at each point in flight. The second is the *guidance mode*, which is understood to be the flight theoretical concept on which the guidance system bases its steering decisions during flight. Finally, there is the control system, which takes the output of the guidance mode and imposes it upon the vehicle. This paper is concerned with only the second of these three items, the guidance mode.

THEORY

Performance and Guidance

For use in the following discussion it is convenient to define two "problem areas." These are the *performance problem* and the *guidance problem*. The performance problem is concerned with selecting the "best" path to follow in order to accomplish some mission. The best path is defined as that path which extremizes some quantity. In the following discussion it will always be assumed that flight time is to be minimized, which is equivalent to maximizing payload in the vehicle systems under study.

The guidance problem involves the instantaneous evaluation of the vehicle state and the generation of a steering program which will lead to mission accomplishment. Also included in the guidance problem is the requirement that the vehicle terminate thrusting at the appropriate time. Thus, the guidance problem is to evaluate the vehicle's instantaneous state coordinates and from these generate a steering program and an estimated time to terminate thrust. If the solution which is generated by the guidance system is identical to the solution of the performance problem at every point, the guidance mode may be said to be optimum.

Guidance Mode

Many different techniques have been used to generate guidance modes in the past. Two of these schemes, which are typical, are the "delta minimum" scheme and the "velocity-to-be-gained" scheme. Each of these methods has its advantages under an appropriate set of circumstances and is described briefly. The delta minimum scheme assumes that the state coordinates at any point in time will only be slightly different from those of some precomputed standard trajectory. This precomputed trajectory is stored on-board the space vehicle. In flight a comparison is made between the vehicle state coordinates and the corresponding values of the precomputed standard trajectory with the difference in the coordinates being used to generate an error signal. The guidance mode acts to null this error signal. Cutoff is given when the vehicle pierces an empirically determined surface in phase space which is computed by curve fitting an area near the standard cutoff value.

In the velocity-to-be-gained guidance mode an instantaneous velocity vector is measured at each point in flight. A vector is computed which, if obtained at that instant, would lead to mission fulfillment. The difference between these two vectors is used as an error signal for steering commands and cutoff.

Each of these two guidance modes has been used in flight vehicles and performed satisfactorily. However, some of the inadequacies of such simplified approaches may be recognized from the following discussion of the need for the adaptive guidance mode.

Need for the Adaptive Guidance Mode

The adaptive guidance mode is under development at the George C. Marshall Space Flight Center in an effort to overcome some of the limitations which are encountered in simplified schemes. The introduction of large, complex, multi-purpose vehicles such as the Saturn leads to severe requirements on the guidance system. No longer is it possible to assume that only one type of mission will be flown. The guidance system must be capable of handling large perturbations in the state coordinates such as those which would be introduced by the failure of one engine of a cluster of engines. The adaptive guidance mode also is independent of the location of the computations. This flexibility might be utilized on a lunar mission, for example, by computing the steering program and cutoff function on-board during boost flight. After termination of thrust of the last stage of boost flight, the location of computation could be transferred to ground facilities with information and command channels linking the vehicle to the ground. For landing on the moon the location of computation might be maintained at the earth, transferred back on-board the vehicle, or possibly transferred to a lunar base, when one is established.

One of the greatest advantages of the adaptive guidance mode is the flexibility which it extends to the designer. The same concept may be used for flights which require very little in the way of guidance and for flights which are extremely demanding of the guidance mode. The adaptive guidance mode can help the designer to measure the influence of hardware constraints on a system. This results from the fact that at each point the theoretical optium solution is known. If an additional constraint is added, the resulting degradation in system performance indicates the cost of that constraint. Thus, the designer has immediate control over the degree of optimization of the guidance scheme and a method of easily performing system optimization studies.

Two Point Boundary Value Problem

The adaptive guidance mode attacks the problems described in the previous section by an instantaneous solution of the performance problem at each point in flight. The problem which is to be solved is a two point boundary value problem in the calculus of variations. We know the state coordinates of the vehicle at any time. We also know a functional description of the end conditions which are desired. Thus, we have conditions which must be met at each of the two boundaries of the problem. The problem then reduces to the selection of that path which satisfies both end conditions and at the same time renders flight time a minimum. In most cases this problem may not be solved in closed form, but rather a numerical solution is necessary. A numerical approach to the two point boundary value problem is to assume values for all of the control variables which are unknown at one of the boundaries. The equations are then numerically integrated to obtain the resulting conditions at the second boundary. An iteration may be used to select values at the first boundary which give the desired results at the second. Thus, whereas the solution is basically one of calculus of variations, it is heavily dependent upon numerical analysis and the exploitation of large computers. Work is in progress which will hopefully lead to a closed solution to the two point boundary value problem. However, until such a closed solution is available, it will be necessary to continue with numerical solutions. This technique is not unreasonable, since an IBM 7090 computer can give the desired solution to a typical Saturn two point trajectory optimization problem in less than one minute. Although this is a short time, it would still be desirable to reduce the time of the numerical solutions, since many such individual solutions are used in the over-all solution to the guidance problem. Effort is being expended in this direction also.

Mission Criteria

In the past it was usually a relatively simple matter to mathematically express the conditions desired at cutoff, the end of the propelled phase of flight. Frequently only one stage was flown with guidance, with the cutoff point being defined to be a circle, perigee of an ellipse, or injection into a ballistic flight (ellipse) having a specified range at impact. However, as missions become more complex, the formulation of mission criteria becomes more formidable. Consider, for example, the flight of a vehicle to impact on the moon. Assume that three stages are used during boost flight. The shaping of the first stage will be largely dictated by the necessity to survive certain atmospheric forces. The second and third stages, however, are available for optimum shaping. In order to shape the second stage in an optimum manner, the guidance mode must look forward to the end condition of the mission, that is the lunar impact, and shape second stage flight in such a manner as to give optimum performance in reaching that point. Clearly the introduction of any other arbitrary intermediate mission criteria at the burn-out points of the individual stages can only lead to performance degradation, since these represent additional, unnecessary constraints on the system.

The attainment of a complete analytical solution to as complex a problem of mission formulation as the one posed on lunar flights will require a great deal of research. Extensive work has been done in celestial mechanics which pertains to this problem, but the field is still wide open for both analytical work and numerical studies. Numerical methods can be applied well, since only solutions within certain regions restricted by practical considerations are needed. These numerical methods, taking full advantage of new, advanced computing facilities, may consist first of generating families of trajectories which satisfy the pertinent two point boundary value problem in celestial mechanics. Next, the region of this family that coincided with the expected cutoff region of the preceding powered phase would be represented in some form by numerical curve fitting procedures. Analytic methods of solution are also aided by the limited region aspect. Also, analytic methods should provide judgement as to which coordinates and which functions are best suited for the numerical procedures.

It might be pointed out that presently available procedures appear to be sufficient to provide the functions required by the adaptive guidance mode. The main object of such research efforts as just described is to economize on the time and expense of obtaining results by present methods, and to minimize the quantity of computations required of the on-board computer.

Steering Function and Cutoff Function

A system of equations that describes a two dimensional vacuum flight in an inverse square gravitational field of a vehicle having constant thrust and mass flow is:

$$\frac{F}{m}=\frac{\left(\frac{F}{m}\right)_0}{1+\left(\frac{\dot{m}}{m}\right)_0(T-T_0)} \tag{1}$$

$$\ddot{X}=\left(\frac{F}{m}\right)\sin\chi-\left(\frac{k^2}{r^3}\right)X \tag{2}$$

$$\ddot{Y}=\left(\frac{F}{m}\right)\cos\chi-\left(\frac{k^2}{r^3}\right)Y \tag{3}$$

At any time in flight, T_0, the state coordinates

$$X_0,\ Y_0,\ \dot{X}_0,\ \dot{Y}_0,\ T_0,\ \left(\frac{F}{m}\right)_0,\left(\frac{\dot{m}}{m}\right)_0 \tag{4}$$

may be assumed to be known. Application of the calculus of variations theory leads to the requirement that

$$\ddot{\lambda}_1=\left[\frac{k^2}{r^5}\right][\lambda_1(2X^2-Y^2)+3\lambda_2XY] \tag{5}$$

$$\ddot{\lambda}_2=\left[\frac{k^2}{r^5}\right][3\lambda_1XY+\lambda_2(2Y^2-X^2)] \tag{6}$$

$$0=\lambda_1\cos\chi-\lambda_2\sin\chi \tag{7}$$

be satisfied together with equations (1–3) and the boundary conditions in order that the χ function extremize $(T-T_0)$. The solution for the function $\chi(T)$, evaluated at $T=T_0$ is

$$\chi_0=\chi_0\left(X_0,\ Y_0,\ \dot{X}_0,\ \dot{Y}_0,\ T_0,\ \left(\frac{F}{m}\right)_0,\left(\frac{\dot{m}}{m}\right)_0\right) \tag{8}$$

The solution of the given system of equations will also lead to an expression for the time of cutoff, T_f,

$$T_f=T_f\left(X_0,\ Y_0,\ \dot{X}_0,\ \dot{Y}_0,\ T_0,\ \left(\frac{F}{m}\right)_0,\left(\frac{\dot{m}}{m}\right)_0\right) \tag{9}$$

The right hand sides of equations 8 and 9 are referred to as the *steering* function and the *cutoff* function, respectively. It is the quality of these two functions which determines the degree to which the performance problem is satisfied at each point. The following section on applications takes up the problem of generating equations 8 and 9. Whereas the equations (2–9) have assumed two dimensional flight, the extension to three dimensions may be readily made. In this case two angles are required to specify the orientation of the vehicle, which leads to the requirement for steering functions in both pitch and yaw.

APPLICATION

Introduction

The adaptive guidance mode has been implemented for a variety of missions. That is, the guidance equations have been developed, and the actual in-flight behavior of a vehicle operating under them simulated. These missions have utilized both two and three stage vehicles for injection of spacecraft into orbital mission with and without rendezvous requirements, into reentry flights as well as into transits to the moon. Effort has primarily been directed toward achieving a good formulation of the mode and an understanding of the problems of implementation rather than producing a large amount of data. Direction of studies on adaptive guidance is centered at MSFC. This direction guides the work of a number of universities and industrial concerns on problems of application and supporting research. Research is being pursued primarily in the fields of calculus of variations, celestial mechanics, and the exploitation of large computers.

Assumptions

One of the missions for which the adaptive guidance mode has been implemented is injection of a Block II, Saturn C–1 vehicle into a 150 nautical mile circular orbit independent of range. Launch was assumed to have taken place from Cape Canaveral, Florida. Due to tracking considerations it was specified that the flight plane of the vehicle, defined at orbital injection, be the plane containing a great circle passing through the launch site at the time of vehicle liftoff at an azimuth angle of 72°, measured east from north. The launch facility requires the vehicle to liftoff on a 100° azimuth. Thus, it is necessary that a roll maneuver be imposed shortly after liftoff to bring the flight into the desired plane at cutoff. The first stage was constrained to fly one fixed, preprogrammed tilt angle history which was selected on the basis of performance for a flight with all conditions nominal. This constraint was imposed due to atmospheric forces and separation considerations.

Steering Function and Cutoff Function Approximation

The success of the adaptive guidance mode has been demonstrated to depend on the quality of the steering equation and cutoff equation shown functionally in equations 8 and 9. The approach which is being taken at this time is to generate these functions numerically by forming approximating polynomials to the true functions. In the example given, a family of first stage trajectories was generated which included the effect of failure of each of the four control engines (one at a time) at 40 seconds of flight time, failure of each of the four control engines at 90 seconds and perturbations in thrust taken about the nominal (all engines operating) case. From each of the first stage end points generated in this fashion, a set of second stage trajectories was calculated. This set of trajectories included variations in second stage mass flow and thrust. All second stage flights were shaped by a three dimensional calculus of variations program which assumed vacuum flight about an oblate earth. Each of these trajectories was required to satisfy the two point boundary value problem defined by the mission criteria at injection and the initial conditions at second stage ignition. In this manner a volume of trajectories was established which covered the region of phase space through which flight of the vehicle might be expected to occur (up to a given level of probability). Each of the points on each of these trajectories then represents a point in space at which the solution to the performance problem is known. That is, at that point the values of χ and T_f are known which are optimum for mission satisfaction. The problem is then to use the known solutions to obtain an approximate solution over the entire volume. This involves curve fitting a function of several variables. No general theory is presently available on this subject. However, good results are being obtained through the use of the least-squares technique. The least-squares technique involves assuming the form of an approximating polynomial. The method then is to minimize the sum of the squares of the differences between the approximating polynomial and the tabulated data points at each point in the volume.

One method of measuring the accuracy of an approximating polynomial is the Root Mean Square (RMS) error. This is defined by the square root of the sum of the square of the differences between the polynomial and the data points at each point in the volume, divided by the number of points used. That is,

$$\Delta\chi = \sqrt{\frac{\sum_{i=1}^{N} (\chi_i - \bar{\chi}_i)^2}{N}} \tag{10}$$

where χ_i = value of χ at the i^{th} point

$\bar{\chi}_i$ = value of the approximating polynomial evaluated at the i^{th} point

N = total number of points used.

Three principal problems have been introduced, selection of the terms in the approximating polynomial, selection of the data points to be used in the curve fitting routine, and estimation of the accuracy of a given polynomial. Whereas it is recognized that the satisfaction of the mission criteria together with minimization of burning time must be the end goal by which the accuracy of a given polynomial is judged, it has been found in practice that the RMS error is a good indication. Therefore, the RMS error was used in this example as the criterion for judging which polynomials should be employed.

Various methods have been tried for selecting the points to be used. Again, this work has mainly been empirical. In this example the data points were selected by sampling each of the family of second stage trajectories at 5 second intervals starting at 140 seconds after liftoff and continuing to 170 seconds. A sampling interval of 40 seconds was applied from 170–610 seconds, after which the interval was decreased back to 5 seconds. This particular set of data points (3474 in number) is referred to as the *statistical* model. The increased sampling around the end points was used in order to increase the quality of the curve fit in those regions. This is especially desired at cutoff, where the mission criteria must be satisfied. The ignition and cutoff points of all of the second stage trajectories lie within the region of dense coverage by data points. In most cases it was necessary to extend the trajectories beyond each of their boundaries in order to provide a good collection of data points for the sample indicated.

The remaining problem was that of selection of the approximating polynomial. This was investigated by assuming a polynomial of the form

$$\chi=\sum A_{hkpqrs}X^hY^k\dot{X}^p\dot{Y}^qT^r\left(\frac{F}{m}\right)^s \qquad (11)$$

where h, k, p, q, r, s were taken to be a set of non-negative integers such that $h+k+p+q+r+s\leq 3$. The limitation to order 3 was arbitrary, but sufficient for the accuracy desired. The term $\left(\frac{\dot{m}}{m}\right)$, which theory indicates should be carried, is not included due to hardware considerations. Studies have indicated only a small decrease in the accuracy of a given polynomial form as a result of dropping the $\left(\frac{\dot{m}}{m}\right)$ terms. This decrease in accuracy may be largely compensated for by the addition of a few more terms to that polynomial.

The cutoff function is generated in much the same manner as the steering function. Results are not yet available on the cutoff function for this particular mission. However, previous studies have proven the feasibility of generation of a cutoff function in the indicated manner. A simplification is introduced by expanding the cutoff function only in the neighborhood (50–100 seconds for example) of expected cutoff. This does not degenerate the guidance mode in any way and may allow more accurate approximating polynomials to be generated.

Results

The Saturn vehicle receives steering commands in a coordinate system, $\overline{X''}$, which is initially fixed with its origin at the launch site. The Y''-axis of the coordinate system is directed parallel to the direction of gravity at the launch point, passes through the center of the earth, and is positive radially outward. The X''-axis is oriented in the firing direction at liftoff. The Z''-axis is then defined to complete a right-hand cartesian system. The orientation of the vehicle axis is specified in a vehicle fixed cartesian system denoted by $\overline{X}_m$. The origin of the vehicle system is assumed to be at its center of mass with the Y axis passing down the long axis of the vehicle, positive through the nose. The X and Z axes are oriented such that they correspond to the X'' and Z'' directions, respectively, when the vehicle is on the launch pad. At any time point the transformation from the inertial system to the vehicle fixed system is given by

$$\overline{X}_m=[\chi_Y]_1\quad[-\chi_R]_2\quad[-\chi_P]_3\quad\overline{X''} \qquad (12)$$

if the assumption is made that the vehicle assumes exactly the commanded attitude. The subscripts on the three rotational matrices indicate the axis about which the transformation was made. For example,

$$[\chi_Y]_1=\begin{bmatrix}1 & 0 & 0\\ 0 & \cos\chi_Y & \sin\chi_Y\\ 0 & -\sin\chi_Y & \cos\chi_Y\end{bmatrix} \qquad (13)$$

Roll is constrained to zero during second stage flight of Saturn vehicles, leaving two angles to be commanded, χ_P and χ_Y. Tables I and II contain typical polynomials for the pitch and yaw steering functions, respectively. In order to determine the contribution to the value of χ of each of the terms, an evaluation for the nominal flight was made at ignition of the second stage. This is presented in Table 27–III for the pitch steering function. The contribution of each of the terms varies over time, dependent on the values of the state variables at the point of evaluation. However, the contributions listed in Table 27–III are typical of those which would be encountered in any of the cases.

A series of in-flight disturbances were assumed and introduced one at a time to determine their effect on injection. The two steering functions indicated in Tables 27–I and 27–II were assumed to be in effect. Cutoff was assumed to have been given at an inertial velocity of 7738 meters per second. The resulting injection accuracies are presented in Table 27–IV.

Greater accuracy, if desired, may be obtained by using more terms in the two steering functions.

TABLE 27-I.—*Typical Pitch Steering Function*

Variable	Coefficient
A_o	$-3.7594 \cdot 10$
X	$-2.7100 \cdot 10^{-5}$
Y	$6.4708 \cdot 10^{-5}$
Z	$-5.8160 \cdot 10^{-5}$
$\dot{Y}$	$1.3616 \cdot 10^{-2}$
$\dot{Z}$	$-1.9121 \cdot 10^{-2}$
F/m	$5.1309 \cdot 10^{-1}$
Y^2	$7.1154 \cdot 10^{-10}$
$Z\dot{Y}$	$-1.6515 \cdot 10^{-7}$
$Y\dot{Z}$	$3.2295 \cdot 10^{-7}$
XY^2	$-2.5843 \cdot 10^{-16}$
Y^3	$8.5879 \cdot 10^{-16}$
$Y\dot{Y}^2$	$5.1123 \cdot 10^{-11}$
$\dot{Y}^3$	$7.9804 \cdot 10^{-9}$
$Y\dot{Z}^2$	$1.6455 \cdot 10^{-10}$
$XY(F/m)$	$-5.7500 \cdot 10^{-12}$
Y^2F/m	$4.3811 \cdot 10^{-12}$
$Y\dot{X}(F/m)$	$1.0253 \cdot 10^{-9}$
$\dot{Z}(F/m)^2$	$5.9772 \cdot 10^{-7}$
X^2T	$2.8873 \cdot 10^{-13}$
XYT	$2.7547 \cdot 10^{-12}$
$\dot{X}\dot{Y}T$	$-1.6542 \cdot 10^{-8}$
$Y\dot{Y}T$	$3.9522 \cdot 10^{-10}$
$Y\dot{Z}T$	$1.1063 \cdot 10^{-9}$
XT^2	$-1.3556 \cdot 10^{-9}$
$\dot{X}T^2$	$6.7291 \cdot 10^{-8}$
$\dot{Y}T^2$	$-6.4193 \cdot 10^{-8}$
T^3	$1.1087 \cdot 10^{-6}$
$X^2\dot{Y}$	$-3.8524 \cdot 10^{-14}$

TABLE 27-II.—*Typical Yaw Steering Function*

Variable	Coefficient
A_o	-5.5754
X	$3.8962 \cdot 10^{-4}$
Y	$3.9333 \cdot 10^{-5}$
Z	$6.2342 \cdot 10^{-4}$
$\dot{X}$	$-3.4305 \cdot 10^{-2}$
$\dot{Y}$	$1.9962 \cdot 10^{-2}$
$\dot{Z}$	$-3.1977 \cdot 10^{-2}$
F/m	$4.9044 \cdot 10^{-2}$
T	$-3.2566 \cdot 10^{-2}$
XT	$-3.0044 \cdot 10^{-6}$
YT	$3.4679 \cdot 10^{-8}$
ZT	$-5.2671 \cdot 10^{-6}$
$\dot{X}T$	$4.3656 \cdot 10^{-4}$
$\dot{Y}T$	$-1.2815 \cdot 10^{-5}$
$\dot{Z}T$	$5.0010 \cdot 10^{-4}$
$F/m\ T$	$-1.7348 \cdot 10^{-3}$
X^2F/m	$-7.3330 \cdot 10^{-11}$
$XY(F/m)$	$-1.2626 \cdot 10^{-11}$
Y^2F/m	$-1.5006 \cdot 10^{-12}$
$XZ(F/m)$	$-2.8128 \cdot 10^{-10}$
$YZ(F/m)$	$-2.2184 \cdot 10^{-11}$
Z^2F/m	$-2.7001 \cdot 10^{-10}$
$\dot{X}^2F/m$	$1.7726 \cdot 10^{-7}$
$\dot{Y}^2F/m$	$-1.8590 \cdot 10^{-8}$
$\dot{Z}^2F/m$	$-3.4311 \cdot 10^{-7}$
$X(F/m)^2$	$6.0921 \cdot 10^{-8}$
$Y(F/m)^2$	$-2.8912 \cdot 10^{-9}$
$Z(F/m)^2$	$1.2572 \cdot 10^{-7}$
$\dot{X}(F/m)^2$	$-2.8831 \cdot 10^{-6}$
$\dot{Y}(F/m)^2$	$-2.2669 \cdot 10^{-7}$
$\dot{Z}(F/m)^2$	$-3.5421 \cdot 10^{-6}$
X^2T	$1.9175 \cdot 10^{-11}$
XYT	$5.5497 \cdot 10^{-12}$
Y^2T	$1.5029 \cdot 10^{-13}$
XZT	$7.2002 \cdot 10^{-11}$
YZT	$9.9162 \cdot 10^{-12}$
Z^2T	$6.8154 \cdot 10^{-11}$
$\dot{X}^2T$	$-6.5189 \cdot 10^{-8}$
$\dot{Y}^2T$	$9.5681 \cdot 10^{-9}$
$\dot{Z}^2T$	$1.1316 \cdot 10^{-7}$
$(F/m)^2T$	$1.7910 \cdot 10^{-5}$
$(F/m)T^2$	$-1.2290 \cdot 10^{-6}$

TABLE 27-III.—*Evaluation of Pitch Steering Function at Ignition of Second Stage for Standard Flight*

Variable	Coefficient	Contribution to χ_p (deg)
A_o	$-3.7594 \cdot 10$	-37.595
X	$-2.7100 \cdot 10^{-5}$	-3.8859
Y	$6.4708 \cdot 10^{-5}$	4.4583
Z	$-5.8160 \cdot 10^{-5}$	2.3827
$\dot{Y}$	$1.3616 \cdot 10^{-2}$	16.093
$\dot{Z}$	$-1.9121 \cdot 10^{-2}$	24.089
F/m	$5.1309 \cdot 10^{-1}$	32.121
Y^2	$7.1154 \cdot 10^{-10}$	3.3778
$Z\dot{Y}$	$-1.6515 \cdot 10^{-7}$	7.9966
$Y\dot{Z}$	$3.2295 \cdot 10^{-7}$	-28.033
XY^2	$-2.5843 \cdot 10^{-16}$	$-.17591$
Y^3	$8.5879 \cdot 10^{-16}$	$.02809$
$Y\dot{Y}^2$	$5.1123 \cdot 10^{-11}$	4.9203
Y^3	$7.9804 \cdot 10^{-9}$	13.175
$X\dot{Z}^2$	$1.6455 \cdot 10^{-10}$	17.995
$XY(F/m)$	$-5.7500 \cdot 10^{-12}$	-3.5564
Y^2F/m	$4.3811 \cdot 10^{-12}$	1.3020
$Y\dot{X}(F/m)$	$1.0253 \cdot 10^{-9}$	10.521
$\dot{Z}(F/m)^2$	$5.9772 \cdot 10^{-7}$	-2.9514
X^2T	$2.8873 \cdot 10^{-13}$	$.89057$
XYT	$2.7547 \cdot 10^{-12}$	4.0827
$\dot{X}YT$	$-1.6542 \cdot 10^{-8}$	-6.9773
$X\dot{Y}T$	$3.9522 \cdot 10^{-10}$	4.8280
$Y\dot{Z}T$	$1.1063 \cdot 10^{-9}$	-14.406
XT^2	$-1.3556 \cdot 10^{-9}$	-4.3744
$\dot{X}T^2$	$6.7291 \cdot 10^{-8}$	3.6025
$\dot{Y}T^2$	$-6.4193 \cdot 10^{-8}$	-1.7074
T^3	$1.1087 \cdot 10^{-6}$	3.7429
X^2Y	$-3.8524 \cdot 10^{-14}$	$-.93618$

TABLE 27-IV.—*Errors at Cutoff Resulting From Application of Guidance Mode*

Error source	Error at cutoff		
Standard first stage with second stage variations	Δ Altitude (km)	Δ Theta (deg)	Δ Inclination (deg)
No Variations	+0.16	+0.008	+.00059
+1% Flow Rate	+0.28	+0.016	+.00064
−1% Flow Rate	+0.06	−0.013	+.00051
+1% Thrust	−0.06	−0.017	+.00059
−1% Thrust	+0.30	+0.032	+.00059
+500 lb Weight	+0.16	+0.010	+.00059
−500 lb Weight	+0.12	+0.004	+.00059
Nominal second stage with first stage variations			
Head Wind	+0.22	+0.002	+.00058
Tail Wind	−0.04	+0.004	+.00059
Right Cross Wind	+0.14	+0.006	+.00059
Left Cross Wind	+0.16	+0.008	+.00059
+1% Flow Rate	+0.12	−0.006	+.00052
−1% Flow Rate	+0.24	+0.016	+.00056
+1% Thrust	+0.12	+0.001	+.00063
−1% Thrust	+0.24	+0.012	+.00059
+5000 lb Weight	+0.22	+0.014	+.00060
−5000 lb Weight	+0.12	−0.002	+.00057
#1 Engine Out at 100 sec	+0.12	+0.014	+.00060
#2 Engine Out at 100 sec	+0.20	+0.042	+.00056
#3 Engine Out at 100 sec	+0.10	+0.028	+.00058
#4 Engine Out at 100 sec	−0.12	−0.002	+.00060

CONCLUSIONS

The development of the adaptive guidance mode has been shown as a consequence of the requirements of advanced space vehicles. The study and implementation of this guidance mode involves special knowledge and research in the disciplines of calculus of variations, celestial mechanics, and the use of computers. General knowledge of almost all fields of mathematics and classical physics is a necessity. Whereas good results are being obtained at the present time, as exemplified in Table 27-IV, it is expected that even better results and more economical methods may be obtained through the introduction of more analytical techniques and further study.

N63-11111

28. A Survey of Midcourse Guidance and Navigation Techniques for Lunar and Interplanetary Missions

By John D. McLean

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INTRODUCTION

This paper is concerned with problems involved in navigation during the midcourse portion of a space mission. Briefly, this phase of the mission is defined as that portion of the trajectory during which the vehicle is in transit between celestial bodies and is not being acted on by large propulsive or aerodynamic forces.

The purpose of the phase immediately preceding the midcourse is to inject the vehicle along a trajectory which will reach the desired destination. It is only because this injection cannot be accomplished exactly that any midcourse guidance is necessary. An example of the need for, and use of, midcourse guidance occurred recently in the case of Mariner II. It was desired to inject the vehicle on a trajectory which would pass within 9,000 miles of the planet Venus, but the small errors inherent in the injection guidance system resulted in a trajectory which would have missed by more than 200,000 miles. A midcourse velocity correction was made subsequently in order to reduce the miss to an acceptable value.

MIDCOURSE NAVIGATION

The operations necessary for the modification of the trajectory are accomplished in three steps. First, data from which the vehicle's position and velocity can be determined must be collected. Second, since the instruments used to collect the data are subject to inaccuracies, it is necessary to use redundant data and statistical processing for a satisfactory estimate of the trajectory. Finally, a velocity correction must be made which will reduce the miss at some selected target point to an acceptable value.

The data collection could be accomplished by either ground-based tracking stations or by instruments aboard the vehicle. Ground-based radar tracking has been used quite effectively for determining the orbits of near-earth satellites and the trajectories of deep space probes. These data consist of various combinations of line-of-sight angles, range, and range rate. On-board instruments, however, would most likely make optical determinations of the position of the moon, earth, or other planets with respect to the star background.

The method used depends on the mission. Because of payload limitations, unmanned missions have been limited to ground-based tracking as the means of collecting data. However, an unmanned interplanetary mission (ref. 1) using an automatic on-board optical system has been proposed and appears to be feasible. For the manned mission the on-board system becomes attractive for two reasons. First, the optical navigation instruments can be operated by the astronaut and thereby can be simplified to provide greater reliability than that of an automatic system. Second, since ground-based data certainly would not be ignored, the redundancy provided by the on-board system greatly enhances the safety of the mission.

The use of optical instruments aboard the vehicle is very similar to the problem of navigating a ship or airplane as shown in figure 28-1. In this case, we have an estimate of posi-

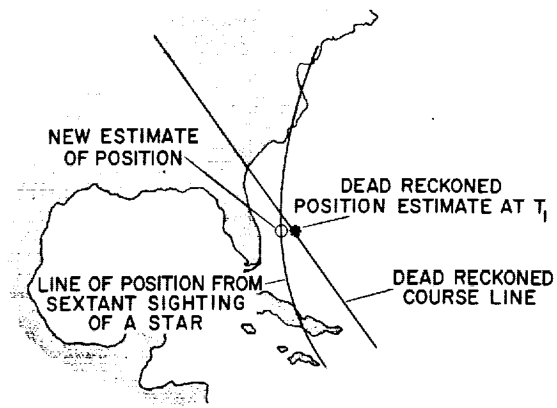


FIGURE 28-1.—Position estimation on earth.

tion at time T_1 on a dead-reackoned course. At this time, a sextant sighting of a star is made and a line of position is drawn through all points from which such a sighting could have been made. This line then traces a circle on the surface of the earth. The point on this line of position nearest the dead-reckoned position is the new best estimate of position. It is interesting to note that if two different stars could be sighted simultaneously and precisely, the intersection of the two resulting position circles would be the exact position.

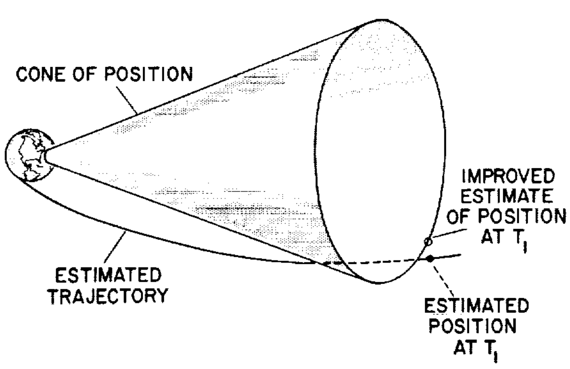


FIGURE 28-2.—Position estimation in space.

Figure 28-2 shows the extension of this process to space navigation. In this case the dead-reckoned course is replaced by the estimated trajectory and a similar estimate of position along the trajectory at a specific time. If now a sextant sighting is made between a star and the center of the earth, and a surface is passed through all possible positions from which the resulting angle could have been measured, a cone is obtained with its apex at the center of the earth. The point on this cone nearest the present estimated position becomes the improved estimate of position. In this case, the position could be determined exactly (assuming perfect instruments) by simultaneous sightings of three stars and the center of the earth. Note that for either ground-based or on-board measurements it is impractical to measure the velocity vector directly; it must be computed from successive position measurements.

Since errors exist in the data collected, it is necessary to process relatively large amounts of data statistically in order to achieve the desired accuracy. One method which has been developed for combining these data (refs. 2, 3, 4) is illustrated in block diagram form in figure 28-3 as follows: Angles between celestial bodies and stars are observed with on-board optical instruments and the raw data are transmitted to the earth computation center to be processed along with ground tracking data.

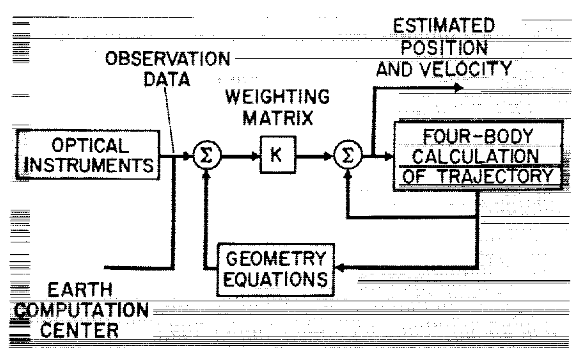


FIGURE 28-3.—Trajectory estimation process.

Initially the best estimate of the trajectory is that one into which it was desired to inject the vehicle. This trajectory is calculated by integrating the vehicle's equations of motion from the set of desired initial conditions. At the time an observation is made the geometry equations are used to compute what the magnitudes of the observed quantities would be if the vehicle were on the estimated trajectory. The differences between the actual observations and those computed are then multiplied by a weighting matrix, K, to give an increment to be added to the estimated position and velocity. The process is then repeated at intervals along the entire trajectory. The matrix, K, is a function of the estimated position and velocity and of the errors in observation. The mathematical procedures, given in the references, for calculating this matrix also provide an estimate of the uncertainties in the knowledge of the trajectory. The estimation procedure essentially provides a six-dimensional equivalent of the minimum mean square error.

The physical interpretation of the process, which is basically the same for manned and unmanned vehicles on either lunar or interplanetary missions, is illustrated in figure 28–4. Here

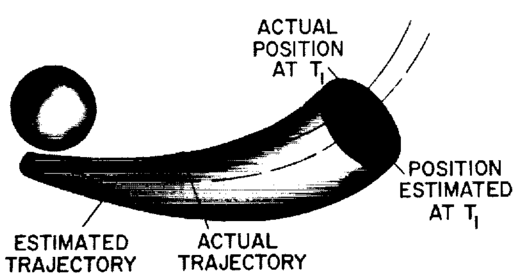


FIGURE 28–4.—Estimation process—trajectory uncertainty.

the situation at and shortly following injection into the transit trajectory is depicted. Since injection cannot be perfect, there is an area of uncertainty, as indicated by the small ellipse, representing the location at injection of all possible trajectories. Actually, a volume of uncertainty exists in the form of an ellipsoid which is reduced to two dimensions for ease of illustration. The size of this ellipse is a function of injection accuracy. After a period of time, this area will grow as indicated by the cone and, at time T_1, the uncertainty is represented by the large ellipse. The actual trajectory is also shown in the figure along with the actual and estimated positions at T_1. At this time a sextant sighting of the earth is made. The cone of position due to this observation is indicated in figure 28–5, and because of errors in measurement, this cone has a thickness associated with it as indicated.

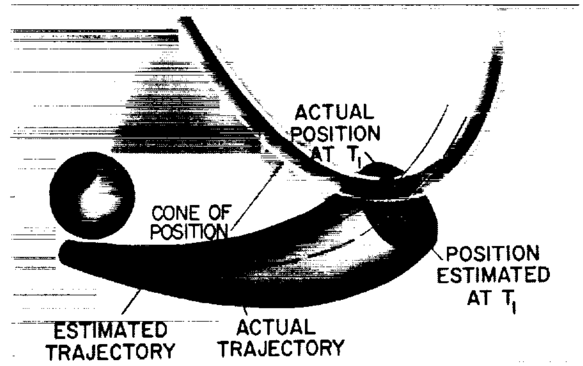
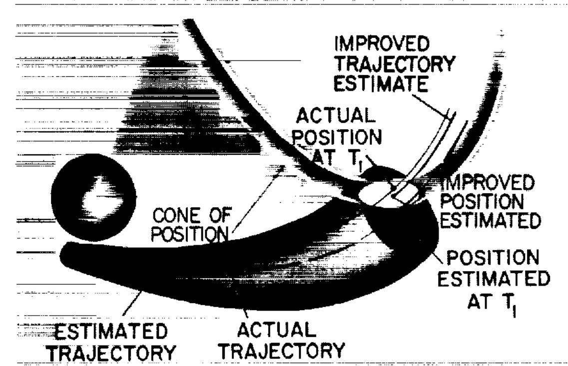


FIGURE 28–5.—Estimation process—measurement uncertainty.

If the two areas of uncertainty, one from the trajectory and one from the cone, are combined statistically, a new ellipse of uncertainty is obtained as shown in figure 28–6. The operations using the weighting matrix mentioned previously place the new best estimate of position at the center of this ellipse. There is a corresponding best estimate of velocity which, when combined with the position estimate, gives the new estimated trajectory. Of course, observations must be made of the moon or other appropriate bodies, in addition to the earth, each with respect to two or more stars so that the area of uncertainty may be reduced in all directions.



FIGUE 28–6.—Estimation process—combined uncertainty.

The data processing has been explained for on-board measurements. Ground-based tracking data also would normally be available and it would be desirable to make use of it. One conceivable way of combining data from the two sources would be to relay the on-board information to the ground to be processed along with that from the earth-fixed stations. The resulting estimated position and velocity would then be communicated to the spaceship to be used as the best estimate. The mathematical details of this process are being studied at Ames. This dual computation procedure has advantages, for, in case of a communications failure, the on-board computations would provide a good estimate of the vehicle's trajectory. For unmanned vehicles, all the data processing is currently done on earth because of the complexity and weight of automatic on-board equipment, but, in theory at least, either method could be used. For interplanetary flight, particularly near a distant planet, more dependence on vehicle-based measurements will be necessary if a close approach, orbiting or landing, is to be successful.

Having used this estimation process to compute the vehicle's trajectory, we must now determine what velocity change is required so that the resulting modified trajectory will pass through the desired end point. Two methods of calculating this correction will be discussed.

As noted previously, the reference trajectory, illustrated for the outbound leg of a lunar mission on figure 28–7, is that trajectory into which it was originally desired to inject the vehicle. The actual trajectory is somewhat different from the reference due to injection errors and would miss the desired terminal point if no corrective action were taken. It is desired to make a velocity correction at point C to reduce the error at the terminal point to zero. One approach would be to integrate the actual trajectory ahead to its nearest approach to the terminal point and determine the miss. Computations would also be made to determine the change in this miss as a result of changing the velocity at point C. The computed velocity correction could then be added to the estimated velocity and the process repeated until an acceptable miss was obtained. The resulting corrective velocity, thus computed, would then be applied to the vehicle. Such an iteration process gives a very accurate corrective velocity but it also requires a large computing capability. Therefore, this procedure is best suited to unmanned vehicles where the computing is done on the ground.

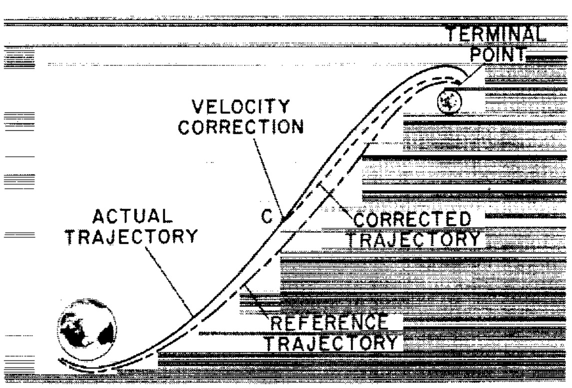


FIGURE 28–7.—Calculation of velocity correction.

Another method, which has been used at Ames because of its suitability to on-board computers, takes advantage of the reference trajectory. This trajectory is computed in advance on the ground and its use depends upon the ability to inject the vehicle into a trajectory which does not differ markedly from the reference. If such is the case the transition matrices between points along the trajectory where velocity corrections are to be made and the terminal point can be precalculated. These matrices are composed of partial derivatives of position and velocity at the terminal point with respect to position and velocity at earlier points on the trajectory. This process is analogous to finding the transfer functions of aircraft by considering small perturbations around trim conditions. The matrices can either be stored for various times along the trajectory or the matrix relating injection to the terminal point can be updated as the mission proceeds. It has been found that this method of guidance is quite accurate for rather large deviations from the reference trajectory.

The problem of when and what observations should be made, and when to make the velocity corrections, remains to be discussed. One schedule of observations and velocity corrections tried in the Ames studies for a circumlunar flight is shown in figure 28–8 along with the assumptions on accuracy and the end results. This particular flight uses 45 observations and 3 velocity corrections on the outbound leg and 35 observations and 2 velocity corrections on the return leg. The observation sequence prior to the first velocity correction is shown in detail. Similar sequences were followed preceding each subsequent correction. This schedule is such that the crew can reasonably be expected to make the observations in a satisfactory manner. The resulting flight requires a relatively small total velocity correction and meets the mission accuracy requirements both at the moon and at the earth on return. Minor alterations of this schedule are permissible and will not significantly change the over-all results.

Basic sighting accuracy, RMS—10 sec of ARC; total velocity correction, RMS—15 M/sec; error in perilune altitude, RMS—3.9 km; error in perigee altitude, RMS—1.4 km.

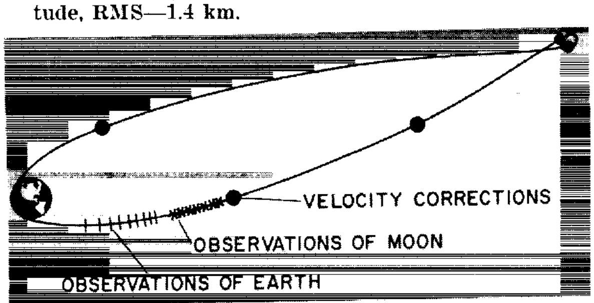


FIGURE 28–8.—Lunar trajectory observation schedule.

ABORT

During the midcourse phase of a manned mission, occurrences, such as solar flares or partial failures of the life support system, may require that the mission be aborted. The type of abort will depend on the equipment still available aboard the vehicle. If all of its navigation equipment is still in operation, the vehicle can navigate during the abort return in the same fashion as in midcourse. If, on the other hand, the primary on-board navigation system has failed, some simpler navigation method will be required to successfully return to earth. In either case, however, the vehicle must still be able to control its attitude and to fire the rocket engines as required.

No matter where the abort occurs, the problem at the time of the abort is to determine a new trajectory which passes through the present position and has a satisfactory perigee similar to the one shown in figure 28–9. This trajectory can be selected so that the fuel available on board is used in the optimum fashion to minimize the return time. The abort indicated in the figure is initiated about 1½ days following launch and the return shown takes another 1.7 days, even though the abort maneuver used all of the fuel that was originally allotted for a lunar landing and takeoff.

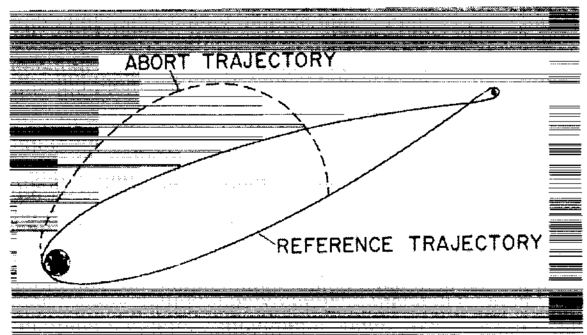


FIGURE 28–9.—Typical abort trajectory.

Merrick and Callas at Ames have shown it to be feasible to reduce the calculation of the abort velocity for minimum time return to a simple graphical operation. This procedure is illustrated in figure 28–10. The necessary radial and normal velocities for return to the proper perigee height are plotted as solid lines for various ranges. The further the velocity increment vector extends to the left along the return velocity curve the shorter will be the return flight time. The return velocity curves

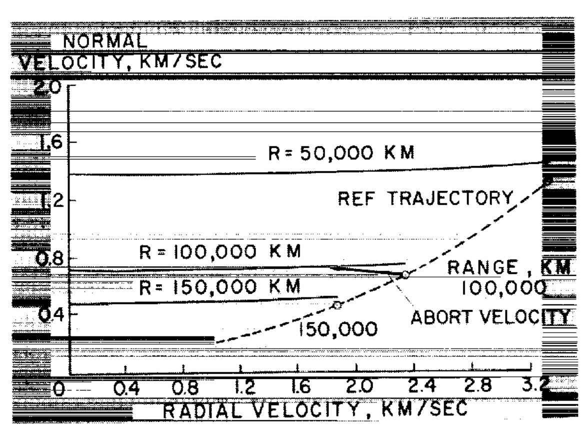


FIGURE 28–10.—Graph for abort computation.

are terminated at the point of minimum incremental velocity. They could be extended to the right until escape velocity is reached, but this region of the curves would represent an increased return time. The velocity associated with the reference trajectory is plotted as a dashed curve. The vector joining the corresponding ranges between the solid and dashed lines is the required velocity increment vector. The velocity corrections calculated in this manner will simply return the vehicle on a trajectory from which a safe entry can be made without regard to the landing site. It is hoped that a similar method can be developed to return the vehicle to a predetermined site if the emergency condition allows.

Since the initial abort maneuver cannot be made perfectly, further corrections will be required. If we assume that the vehicle has had no equipment malfunctions, these corrections would be computed by means of the midcourse guidance technique. These corrections will require less than 10 percent of the fuel used for the initial abort maneuver.

If the vehicle has had failures of the navigation equipment, such as the computer, the sextant, or the inertial platform, the crew must resort to one of two emergency modes of navigation. If communication with earth is still available, ground-based tracking and computational facilities can be used. Under these circumstances, ground facilities will perform the entire navigation computation and simply instruct the crew when to make a velocity correction, how large it should be, and in what direction. If communication with earth is also lost, the crew may have available an emergency navigation procedure which makes use of rudimentary equipment and greatly simplified computations. The equipment might consist of only a camera to photograph the earth and moon against the star background, a scale for measuring distances on the photograph, and charts and tables. A slide rule or small desk calculator might also be provided so that computations could be made manually. Early research by Dewey Havill at Ames on a back-up navigation method of this kind has shown promise but much remains to be done to demonstrate its feasibility.

CONCLUSION

A brief survey has been given of techniques being used or being considered for use during the midcourse portion of space missions. The examples presented are for a primary guidance system and emergency methods which might be used for a manned lunar mission. The techniques are reasonably simple, allow the crew flexibility in the operational sequence, and would allow the use of ground data, if available. Although much work remains to be done on the guidance details for a manned lunar vehicle, enough results have been accumulated to indicate that the remaining problems are not insurmountable. A brief consideration of manned interplanetary flights indicates a midcourse guidance problem which is similar to that of the lunar mission. It also appears that the guidance will be considerably more difficult than for the lunar mission, particularly with respect to emergency procedures. It is reasonable to expect, however, that experience with the lunar mission will point the way to the solution of these problems.

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N63-11112

29. Guidance and Navigation Aspects of Space Rendezvous

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JOHN D. BIRD, *Head of the Astromechanics Branch, Theoretical Mechanics Division, NASA Langley Research Center, received Bachelor of Science degrees in mechanical engineering and aeronautical engineering in 1941 and 1942, respectively, from the Georgia Institute of Technology. He earned his Master of Science degree in aeronautical engineering from the University of Virginia in 1957 through the Langley graduate study program. Bird, who joined the Langley Staff on June 1, 1942, is an authority on the dynamics of airflow patterns about mutually interacting bodies and airfoil surfaces, with an exceptional understanding of both the theoretical and physical aspects of such interference flows. During the past years he has established a reputation as an expert in the highly specialized and complex area of space flight mechanics including the fields of reentry dynamics, rendezvous, and the manual control of docking operations. He has written definitive papers on the dynamics of spinning missiles and reentry bodies, the mechanics of rendezvous, and the lunar orbit rendezvous concept which has been adopted as the primary mode of early lunar exploration. He conceived and developed a tuft grid to study flow fields by visual methods which has been used throughout the nation. He serves as Langley's technical consultant to Marshall Space Flight Center and industry on orbital assembly techniques, launching, and space flight. Bird is a member of the American Rocket Society and is an Associate Fellow of the Institute of the Aeropsace Sciences.*

MANUEL J. QUEIJO, *Head of the Rendezvous Analysis Section, Theoretical Mechanics Division, NASA Langley Research Center, received his Bachelor of*

Science degree in mechanical engineering "with distinction" from Worcester Polytechnic Institute in 1944. He earned his Master of Science degree in aeronautical engineering from the University of Virginia in 1954 through the NASA graduate study program. He has completed course work leading to a doctorate from the Virginia Polytechnic Institute. He is a member of the American Rocket Society and Sigma Xi honorary fraternity. Queijo, who entered on duty at Langley June 12, 1944, has undertaken important problems concerned with lunar landing techniques and control of space vehicles in rendezvous maneuvers. He has completed a study of the piloted phase using simulation equipment and made important contributions in analysis of the various types of landing trajectories which may come into use in lunar landing operations. He is author or co-author of more than 35 technical publications on research he has conducted at Langley. He has done important and original work in the calculation of the loadings on swept wings in sideslip, and designed and developed the only known apparatus for the measurement of oscillatory stability derivatives in pure yawing motion. He designed a "pebble heater" for a Mach 15 heated wind tunnel for dynamic stability work.

SUMMARY

A review is given of the various guidance problems and techniques that are involved in space rendezvous. Along with general studies, specific attention is drawn to the problems pertinent to both earth-orbit and lunar-orbit rendezvous.

Navigation aspects of rendezvous that would be involved, for example, in a manned lunar-landing mission are considered, especially with respect to the establishment of lunar orbits and the constraints that specific landing sites have on rendezvous possibility and on possible return trajectories. Because of the close association with rendezvous-type operations, some of the guidance problems of lunar letdown and of perturbation effects on lunar orbits are also considered.

INTRODUCTION

Rendezvous in space, involving, for example, the ascent of one space vehicle so as to meet with another vehicle already in orbit, has been of widespread interest recently, both scientifically and publicly. Uses envisioned for rendezvous are many, and in fact, several of our nation's major space missions now under development involve rendezvous as a central ingredient.

Much work has been published on rendezvous. Reference 1 contains a long list of published material for reference purposes, and in itself is a broad review of certain basic areas of work that has been done to advance the fundamental understanding of the problems of rendezvous. This reference deals mainly with the aspects of launch and ascent schemes, launch windows, terminal guidance, and braking, and indicates some of the main benefits to be gained by use of rendezvous. The purpose of the present paper is to touch broadly on the guidance and navigation aspects of the space-rendezvous problems. The presentation and results will be given in terms of the lunar-orbit rendezvous scheme for performing a manned lunar-landing mission because this scheme is of high interest and illustrates problem areas quite well. In this connection it should be kept in mind that the problem of going to and landing on the moon is a rendezvous problem of the first order in itself.

To set the stage, the first part of the paper quickly reviews the main aspects that are involved in executing rendezvous from and about a single celestial body. The main part of the paper then deals with the navigation problem of rendezvous when two major celestial bodies are involved. Items considered include such things as launch constraints, coasting-orbit delays, midcourse corrections, conditions necessary to establish lunar orbit, stay times on the lunar surface as affected by rendezvous and return considerations, and perturbation effects on a mother ship in orbit around the moon awaiting the return of a lunar excursion vehicle.

The paper concludes with a brief description of some of the simulators and facilities that are to be involved in continuing studies of rendezvous guidance problems.

REVIEW OF THE MAIN PHASES OF THE BASIC RENDEZVOUS PROBLEM

To set the stage for later discussion in this paper, a quick review is first given of the fundamental steps that are involved in executing a rendezvous from, say, the surface of the earth to an orbiting station. Three main phases are involved: ascent, terminal guidance, and docking.

The ascent phase may be accomplished by either of two basic ascent techniques: direct ascent or the use of intermediate orbits, as illustrated in figures 29–1 and 29–2. In the in-plane direct-ascent scheme shown on the left in figure 29–1, the orbit of the target vehicle is chosen

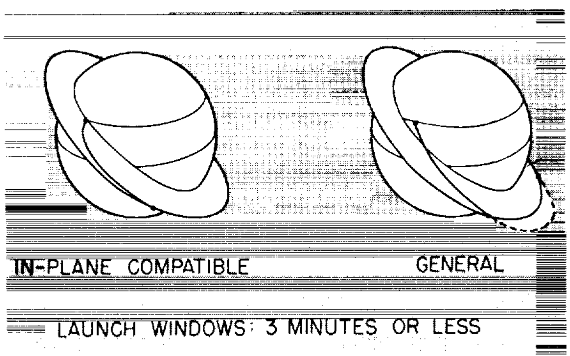

FIGURE 29–1.—Direct ascent.

and maintained by station keeping so that the target passes over the launch site at least once every day; it is during one of these passages that the ferry may be launched for rendezvous—in effect a launch in the orbital plane of the target. On the right of the figure the general direct-ascent scheme is depicted. The ferry is launched when the launch site is close to the orbital plane of the target, executes a dogleg or plane-change type of maneuver, and makes contact with the target either on the outgoing leg of the trajectory or on the return leg. Studies have indicated that the time interval or launch window available for launching and producing a rendezvous by these direct-ascent methods is something on the order of 3 minutes or less, for launch vehicles of present-day capabilities.

To increase the interval of time available for launch so as to be able to absorb holddowns, the intermediate-orbit schemes of figure 29–2 may be used. As seen in the lower left of this figure, if the target orbit has an inclination just slightly greater than that of the launch-site latitude, then the launch site will be close to the target's orbital plane for as long as 3 to 4 hours. The ferry may thus be launched into the orbital plane of the target, without excessive fuel penalty, any time it is ready during this 3- to 4-hour period, irrespective of where the target is. Instead of going into full target orbit, however, the ferry is put into either the chasing orbit shown at upper left or the parking orbit shown at upper right in figure 29–2. Because of the shorter time periods of the chasing or parking orbits, angular deficiencies between the target and ferry are then made up. Next, the ferry is put into coincidence with the target by means of an impulsive-type correction for the chasing-orbit case or an orbital transfer for the parking-orbit case. It should be noted that the main expense in these schemes is time only (and possibly engine restarts); fuel consumption is very efficient.

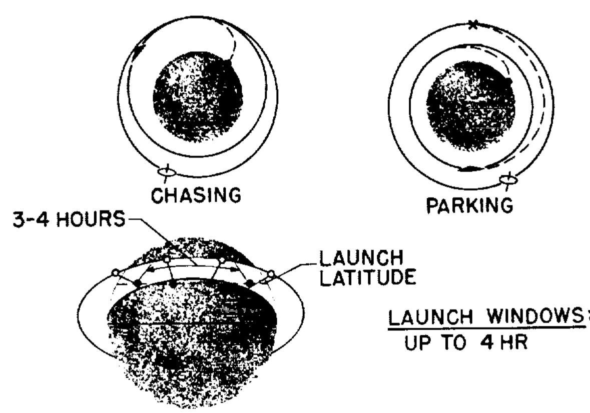

FIGURE 29–2.—Intermediate orbits.

For the terminal phase, the phase beginning when the ferry is about 100 miles or less from the target and wherein the ferry may be guided, or "driven," up to the target, two basic schemes are available (fig. 29–3). In the proportional-navigation scheme the intent is to null the angular motion of the line of sight so that the approach to the target is essentially a one-

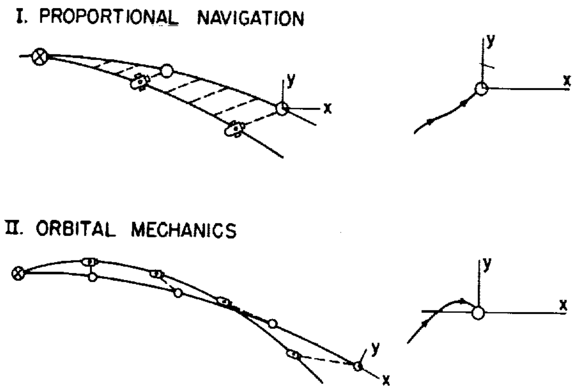

FIGURE 29-3.—Basic terminal-phase schemes.

dimensional motion, and braking (or accelerating) to a soft contact is accordingly then used. In the orbital-mechanics scheme readings are taken to see whether the coasting paths being followed will lead to interception, and if not, propulsion efforts are applied so as to achieve coasting orbits according to the laws of orbital mechanics which do result in rendezvous. (In this scheme a final velocity correction must, of course, be applied just before contact to make speed and direction coincident.) These terminal-phase schemes have been investigated thoroughly, analytically and by means of simulators, for both piloted and automatic systems and for on-off and continuous types of thrusting. The results indicate that the terminal-phase control may be accomplished efficiently by either manual or automatic means, and it is believed that man will demonstrate by far the best performance in executing the actual docking maneuver.

Figures 29-4 illustrates the nature of studies made to determine terminal-phase capabilities of man with a minimum of instrumentation. Using visual cues, the pilot first adjusts his altitude and fires his rockets so that his target becomes motionless with respect to the star background, thereby insuring a collision course. He then simple uses range (R) and range-rate ($\dot{R}$) instruments to go into a braking (or thrusting) sequence to complete the rendezvous. Studies using a simulator shown substantially in Figure 29-4 indicate that this scheme works most effectively.

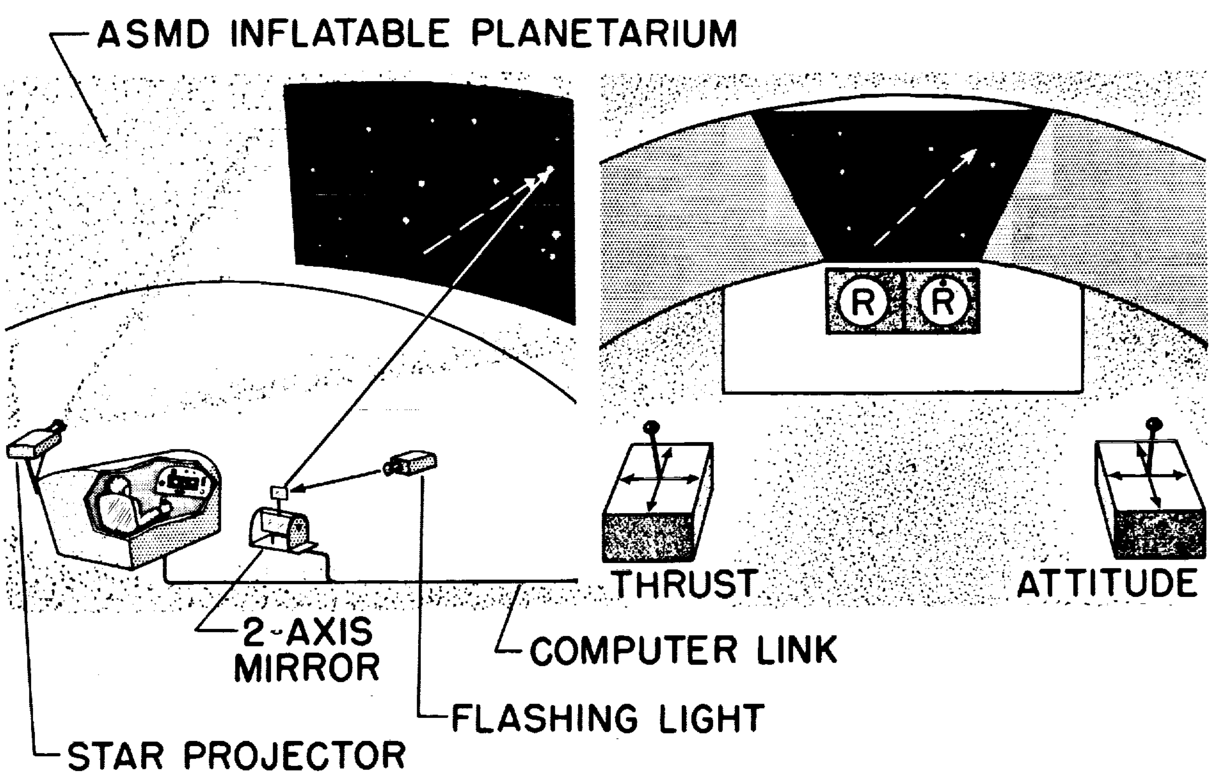

FIGURE 29-4.—Visual terminal-phase simulation.

Figure 29–5 shows the nature of some of the problems and the scope of the present coverage. With respect to the three fundamental aspects of the general navigation problems—(1) where

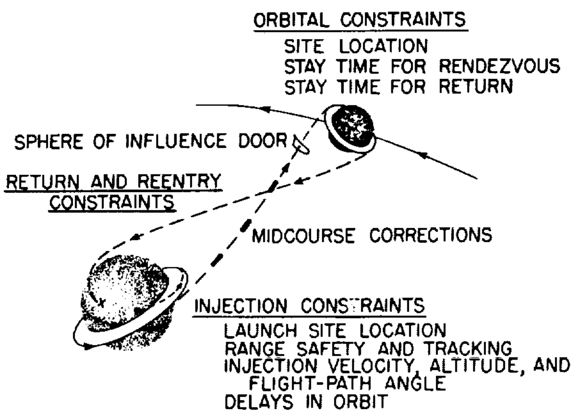

FIGURE 29–5.—Guidance and navigational considerations.

are you, (2) where are you going, and (3) what corrective maneuvers do you make to make yourself go where you want to—most emphasis is given to the second and third items. Included in the considerations are such items as: conditions that affect launch-site location, range safety, and injection parameters required to reach the moon; penalties associated with delays in coasting orbits; the use of different midcourse correction techniques; the establishment of a lunar orbit; the influence of site location on stay time for rendezvous and return-trajectory capability; the perturbation effects on the mother ship remaining in lunar orbit; and the use and advantages of equal-period lunar letdown orbits for the lunar ferry.

Launch Constraints

Some of the factors which must be considered in designing a lunar mission are examined now in a little detail. As was mentioned previously, it is well to remember that reaching the moon is itself basically a rendezvous problem. From figure 29–2 it was noted that the use of a parking orbit increases the available launch window. The following discussion, based on the work of Tolson in reference 3, therefore assumes that both a parking orbit and a coasting orbit around the earth will be features of the lunar mission. These orbits should, of course, be in the desired plane of the vehicle trajectory to the moon. With reference to figure 29–6, the orientation of the final coasting orbit relative to the earth-moon plane is determined by the launch azi-

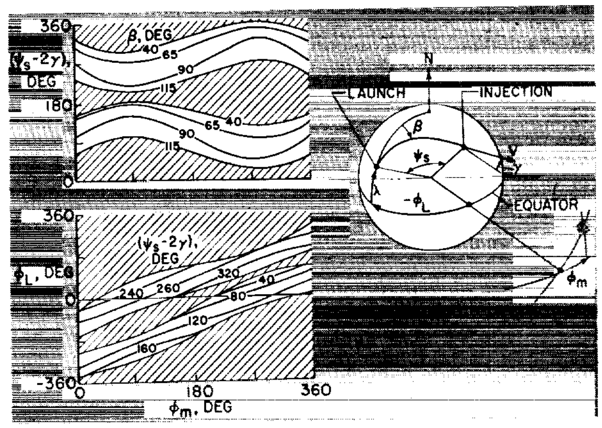

FIGURE 29–6.—Launch constraints. $V/V_p=0.995$.

muth β, the latitude of the launch point λ, and the position of the moon ϕ_m relative to the ascending node.

For illustrative purposes it is assumed that launch will be from Cape Canaveral, which is located at about 28.5° North latitude, and that the coasting-orbit altitude is to be 300 statute miles. The launch azimuth is between 40° and 115° as specified by Atlantic Missile Range safety requirements. The first question of interest is related to the geometric relationship, or the position of the launch and injection points relative to the position of the moon. This again depends on a number of factors, including injection velocity, injection inclination angle, and so on. If an injection velocity of 0.995 local parabolic velocity is assumed, then the quantity $\Psi_s-2\gamma$, involving the geometric angular travel Ψ_s from launch to injection and the flight-path injection angle γ, is related to lunar positions ϕ_m as shown in figure 29–6, in which the parameter β is the launch azimuth angle. As shown for a given launch azimuth, the moon can be reached at any particular position ϕ_m by use of either a long coast (arc of about 300°) or a short coast (arc of about 120°). The two coasting arcs result in different inclinations of the trajectory plane relative to the earth-moon plane. Although not shown here, the angle between the planes can be kept reasonably small (less than 20° throughout the month) by utilizing a long coast when the moon is descending (ϕ_m near 180°) and using a short coast when the moon is ascending (ϕ_m near 0° or 360°).

The required angular positions of the launch point can be determined as a function of the total vehicle angular travel, launch-point latitude, and lunar position, and this is also shown in figure 29–6. As an example of the use of figure 29–6 to obtain approximate initial conditions, consider a launch trajectory with a burning arc of 20°, a coasting arc of 50°, and an injection angle of 15°. For this system,

$$\Psi_s - 2\gamma = 20 + 50 - 2 \times 15 = 40°$$

If the vehicle is to approach the moon at maximum negative declination ($\phi_m = 270°$) the required launch-point location is about $\phi_L = 60°$. It appears, then, that if the launch system is designed so that the coasting-arc length and the launch azimuth can be varied throughout the day, considerable freedom is allowed in choosing the daily launch time also.

Delays in Coasting Orbits

In designing the initial space orientation of the coasting orbit, allowance must be made for orbit precession during stay in orbit, vehicle checkout, preparation for injection toward the moon, and so forth. For nominal conditions of coasting-orbit establishment and anticipated delays, a predetermined time can be established for injection into the lunar trajectory. This predetermined time will be so chosen that the injection velocity is applied in the plane of the parking orbit and the vehicle will arrive at the desired position relative to the moon in a specified transit time. The question arises as to what penalties exist in the event of off-nominal situations when unexpected delays occur. It is clear that the precession of the parking-orbit plane will require modification of the lunar-trajectory injection conditions. This problem has been studied by Wells in reference 4, and a summary of the pertinent results is shown in figure 29–7.

It is assumed that the inclination of the parking orbit is 30° and that the orbit altitude is about 315 statute miles. Because of the oblate-

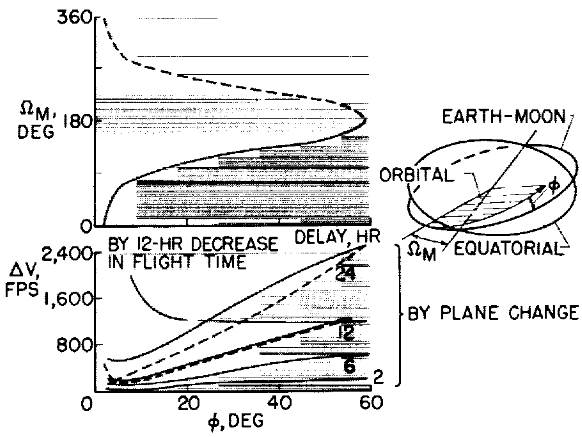

FIGURE 29–7.—Make-up for orbital delay. $\delta_M = 25°$.

ness of the earth, the coasting orbit will precess around the equatorial plane at a constant rate. The equator and the earth-moon plane are inclined at an angle δ_M relative to each other; therefore, the node defined by the intersection of the coasting orbit and the earth-moon plane experiences a regression along the earth-moon plane at a rate which is not constant. The value of Ω_M as a function of ϕ is shown in figure 29–7 for an angle of $\delta_M = 25°$ between the earth-moon and equatorial planes.

Any time spent in the coasting orbit beyond the nominal time will require that a change be made in the trajectory. This could be either a trajectory-plane change or an increase in injection velocity to decrease the flight time necessary.

The cost of each of these methods in terms of characteristic velocity ΔV is shown in figure 29–7 for various injection-delay times; the figure applies for δ_M of 25° and a 60-hour nominal transit time. Plane changes were assumed to be initiated at about 60,000 miles from the center of the earth. The solid and dashed curves indicate negative and positive lunar declinations, respectively. The results for the scheme employing a decrease in flight time are given in the figure for a 12-hour delay period only, since other delays give similar trends. The plane-change technique appears to result in a considerably lower velocity increment, especially for small values of ϕ.

If delay times of several days are considered, other interesting results are brought into view.

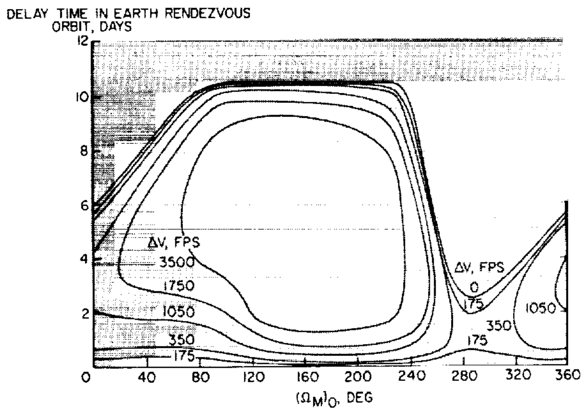

FIGURE 29-8.—Summary of 1968 launch windows.

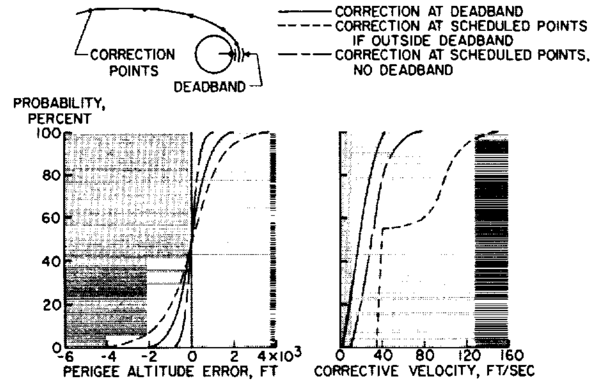

FIGURE 29-9.—Accuracy and corrective velocity for various guidance methods.

Figure 29–8, which is based on the plane-change technique, gives the delay time in days available for a given velocity increment, for each nominal position of the moon. A striking feature of these curves is that at a given Ω_M there are two delay times available for a specified ΔV. Also, there are available combinations of delay times and Ω_M which require no additional velocity increments. Further, with $\Omega_M = 285°$ delay times up to about 3 days can be accepted with only a relatively small additional velocity increment.

Efficiency of Various Midcourse Correction Techniques

The next guidance phase of interest involves midcourse corrections on the way to the moon. Although midcourse correction maneuvers are the specific subject of another paper in this compilation, it seems appropriate to give brief mention to the subject here. Actually much study has been made of midcourse guidance, and notable references exist; individuals who have contributed substantially to the problem include Battin, Schmidt, and Friedlander.

The brief results given here apply to different schemes for making midcourse corrections on a return trajectory from the moon to the earth so as to control perigee altitude of reentry, but similar results may be expected to apply on the earth-moon transfer phase. Figure 29–9 shows results obtained by White (ref. 5) for three methods of control: corrections applied at scheduled points, with and without a perigee deadband, and corrections applied when the predicted perigee points fell outside the deadband. (Random errors were assumed in the measurement of velocity and flight-path angle and in obtaining the desired thrust impulse.) For the results shown in figure 29–9, the initial position of the vehicle was 160° from perigee, and the standard deviations were 1.3 percent for the corrective velocity, $r/10{,}000$ feet per second for velocity, and $0.125r/10{,}000$ degrees for flight-path angle (where r is distance in miles from center of earth to the approach vehicle). The results indicate that the method of applying corrective control when the predicted perigee reached the border of the deadband was slightly less costly than applying control at scheduled points. However, better perigee-altitude control was achieved when the deadband was omitted and corrections were made at prescheduled points.

Lunar-Orbit Establishment

Current studies of manned lunar missions include the establishment of a lunar orbit. For the most efficient establishment of a circular lunar orbit, the lunar-orbit plane must coincide with the selenocentric hyperbola and the orbit must be established at the periselenian point on the hyperbola. Under these conditions the geometric characteristics (inclination, nodal position, and altitude) of the lunar orbit are the same as the selenocentric hyperbola from which the orbit is established. Some studies by R. H. Tolson have examined this situation to define the range of orbits that may be established on this basis. The results of these studies indicate that lunar orbits with a wide variety of geometric characteristics can be established from transfer trajectories for which all the initial earth injection conditions are identical except two, the position of the nodal line of the trajectory and earth-moon planes, and the angular position from this nodal line to the point of injection. It was found that under these conditions all the lunar orbits that could be established tended to have orbital planes with a common line of intersection, and that to a good first approximation this line of intersection defines an entry point on the sphere of influence which would lead to normal lunar impacts. The locations of these entry points are given in figure 29–10 for various inclinations of the lunar transfer trajectory and various injection-

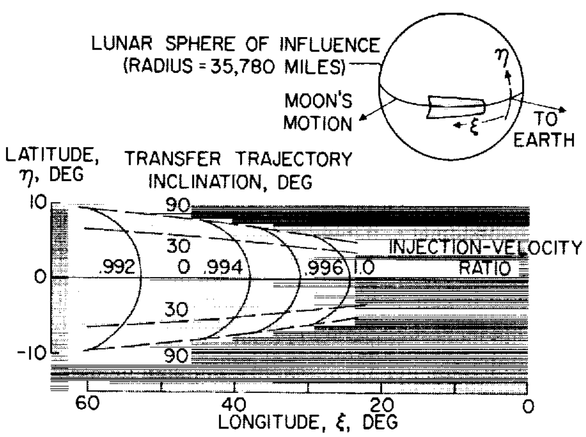

FIGURE 29–10.—Location of entry points on lunar sphere of influence.

velocity ratios. A good approximate expression tying together uniquely the inclination and nodal position of the lunar orbit in terms of these entry points is shown in figure 29–11, which also shows specific results for the following injection conditions; velocity equal to 0.995 parabolic velocity, flight-path angle of zero, trajectory-plane inclination relative to earth-moon plane of 30°, injection-point radius of 4,259 miles, and lunar-orbit radius of 1,180 miles. It is to be noted from this figure that a substantial range in lunar-orbit inclination is available, down to as low as 4°, without any plane-change maneuver being required. As mentioned, the actual value of the inclination depends on the point of injection from earth

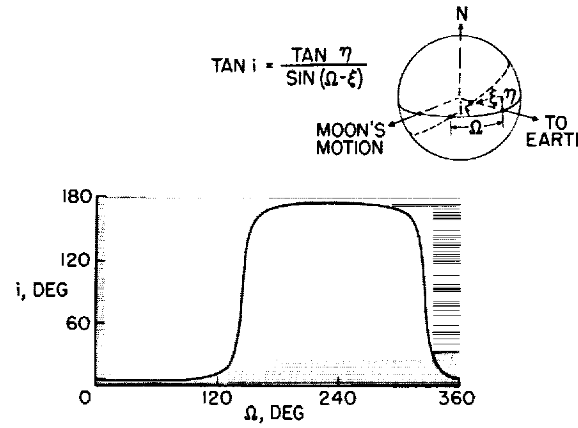

FIGURE 29–11.—Lunar-orbit establishment.

orbit; of significance is the fact that once the inclination is chosen, the nodal position also becomes fixed.

Lunar Landing Sites and Stay Times

When the relationships between the lunar-orbit characteristics and the earth injection parameters are known, it is possible to consider the allowable stay times on the lunar surface as defined by lunar-orbit rendezvous considera tions. An analysis of this matter was made by W. H. Michael and R. H. Tolson. Some of the important parameters of such an analysis are shown in figure 29–12, which represents a schematic of the moon. Suppose it is desired to land at some latitude on the surface. One method of doing this is to establish a lunar orbit with an inclination i greater than the required latitude by the offset δ. When the exploration vehicle starts its descent to the lunar surface,

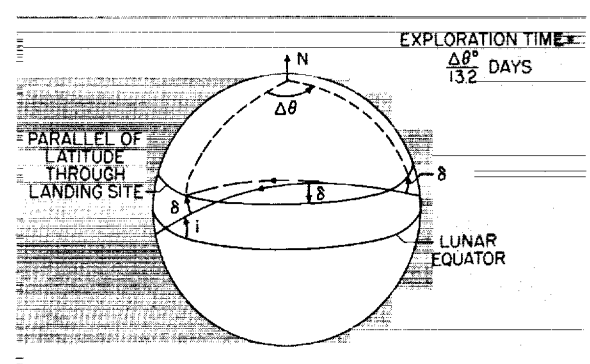

FIGURE 29–12.—Exploration time from rendezvous considerations.

a small out-of-plane impulse is applied so that the path is along the dashed line and the landing site is out of the orbital plane by an angle δ. Because of the moon's rotation on its axis, there will be a relative motion between the landing site and the orbital plane. On the figure the landing site will appear to move to the right along a parallel of latitude of about 13.2° per day. After a certain angular travel $\Delta\theta$, the landing site will again have an offset of δ and the "return to orbit" phase of the mission must be initiated so that a specified fuel expenditure is not exceeded. The exploration period on the lunar surface, which is a function of the landing-site latitude, is given in the upper right-hand corner of figure 29–12, and at any time during this period the vehicle can return to the orbiting satellite with not more than an offset of δ required.

Now the significance of the previously derived relationship between orbital inclination and nodal position is realized. For if δ is specified from propulsion considerations, the latitude of the landing site determines the required orbital inclination through the relation Inclination = Latitude + Offset. But, as shown previously, for a specified transfer trajectory the nodal position is determined once the orbital inclination is chosen. Hence, if this type of landing maneuver is utilized, the longitude of the landing-site location on any parallel of latitude is uniquely determined by the transfer-trajectory characteristics and the offset δ.

The allowable exploration time at each landing site will be different, and perhaps a better way to approach the problem is to ask at what point on the lunar surface the vehicle can land and stay a specified time without requiring a landing offset of more than δ. These points can be determined by finding the position on a parallel of latitude at which, if the vehicle lands there, it will, after the specified time, have an offset δ. For this approach, figure 29–13 shows typical landing-site-time restrictions from these rendezvous considerations for a low-inclination, medium-energy earth-moon transfer trajectory. (These boundaries can of course be shifted by appropriate alteration of the earth injection conditions.) The boundaries shown in the figure indicate the regions of the lunar surface

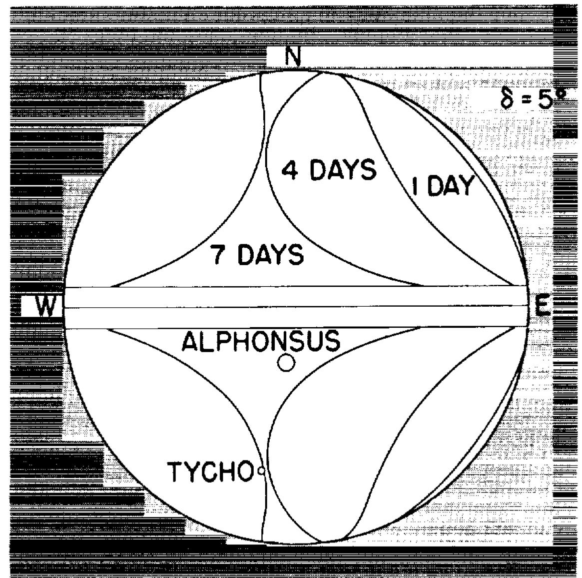

FIGURE 29–13.—Landing-site-time restrictions from rendezvous considerations.

at which landings can be made and at which the vehicle can stay for the specified time without requiring take-off and landing offsets of more than 5°. In the equatorial and polar regions the vehicles can stay on the surface indefinitely and still have the capability of returning to the mother ship within the limiting amount of offset. It is seen that a considerable portion of the lunar surface is subject to manned exploration for periods of a few days or less. Even though there is a relationship between lunar-orbit inclination and nodal position, a lunar orbit can always be established which passes over any arbitrary point on the lunar surface. Hence, an orbit can always be established which passes within δ of any arbitrary point; however, the inclination of the resulting orbit may not be "latitude plus offset." In this case, the exploration is limited by the time it takes for the landing site to move from δ on one side of the orbital plane to δ on the other side. Regardless of the landing-site position, this procedure assures that any point on the lunar surface can be explored for $\delta/6.6$ days; that is, a 5° landing and return offset assures exploration times of about 18 hours at any point on the surface.

So far, the question of landing sites and exploration times has been considered from the standpoint of effecting rendezvous from the lunar surface to the command module. Another consideration is the question of returning to the earth from this lunar orbit. Reference 6 indicates that the specification of reentry constraints leads to the conclusion that the return-to-earth capability is strongly dependent on the inclination and nodal position of the lunar orbit from which the return trajectory is to be initiated. An analysis is in progress to show how these return requirements will affect the curves for landing site and exploration time in figure 29–13. A preliminary result of this study is shown in figure 29–14, where landing sites along

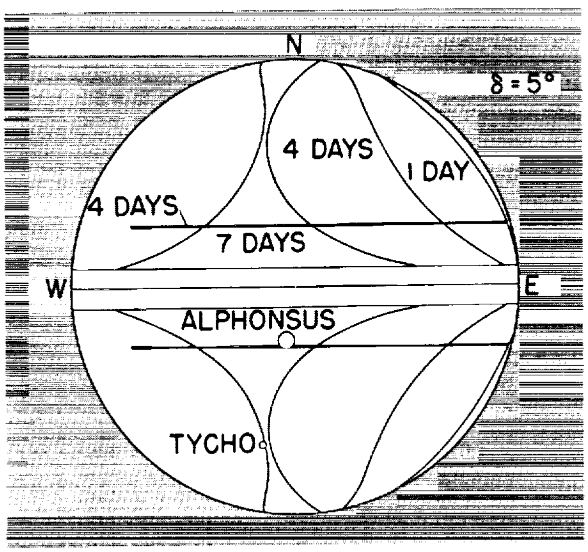

FIGURE 29–14.—Landing-site-time restrictions from return-trajectory considerations.

the 15° parallels and stay times of 4 days are considered for which, during the entire exploration time, the mother ship has the capability of initiating a proper earth-return trajectory without making orbital-plane changes and with a total return flight time of less than 100 hours and a total velocity expenditure of less than 3,100 feet per second. The study indicated that the vehicle can satisfy these requirements by landing anywhere along the heavy horizontal lines. If these lines are compared with the curves obtained from rendezvous considerations, it is seen that in certain regions on the western limb the exploration time may be "return-capability limited" while on the eastern limb some regions will be limited from rendezvous considerations. In the central region the landing sites are compatible with both requirements.

Lunar Landing, Abort, and Take-Off Operations

Another important matter in lunar rendezvous involves the use of equal-period orbits as illustrated in figure 29–15. Consider the situa-

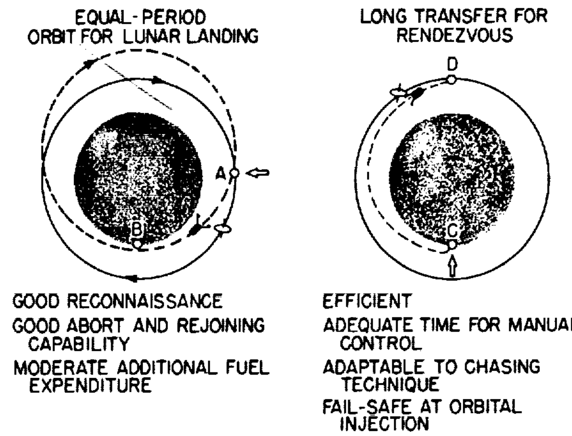

FIGURE 29–15.—Lunar landing, abort, and take-off operations.

tion that exists in the lunar-orbit rendezvous concept when a lunar-excursion module descends from an orbiting command module for a lunar landing. Use of the equal-period orbit requires deflecting the flight path downward at point A so that the landing vehicle descends to a very low altitude just prior to braking for a landing at point B. The energy, and hence the period, of the lunar orbit is not changed in this operation. Use of this technique enables low-altitude reconnaissance to be made on, say, one orbit prior to landing on the next, and in addition provides for rejoining the vehicle left in orbit at the point A on each successive orbit if the reconnaissance is unfavorable or if some fault in the landing engine is detected. The equal-period orbit requires only moderate additional fuel expenditure over other letdown techniques.

In the take-off and rendezvous operation necessary in the lunar-orbit rendezvous concept, there are several points in favor of the use of a long transfer from launch to final rendezvous. Such a procedure is efficient in fuel consumption, and it provides adequate time for manual control or monitoring of the rendezvous guidance functions. In addition a long transfer is adaptable to the chasing technique in the event of a delayed launch. In the chasing technique, injection velocity is withheld at the nominal rendezvous point, D, so that the ascending vehicle moves in an elliptic orbit of lower energy and shorter period than the orbit of the command module. As a result the ascending vehicle overtakes the orbiting vehicle after several orbits, and then rendezous is completed, as was discussed in connection with figure 29–2.

Also, the long transfer orbit has a fail-safe feature in the event of rocket-ignition failure at the rendezvous point D. In this situation the ascending vehicles travels in an orbit which has a perilune altitude equal to the altitude of thrust cutoff at launch, point C. The escending vehicle may hold in this orbit until the difficulty is rectified, with no danger of intercepting the lunar surface as would be the case if a short transfer orbit is used.

Lunar-Orbit Perturbations

Another problem that exists is the effect of perturbations on the lunar orbit due to oblateness effects. These effects influence the rendezvous problem at the moon and must be taken into consideration. This problem has been studied by R. H. Tolson. Figure 29–16 illustrates the effect involved for the case of a satellite in a mean circular orbit having an altitude

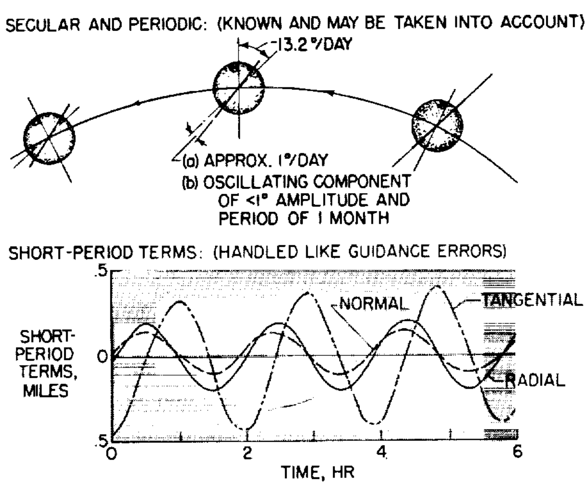

FIGURE 29–16.—Perturbations of a close lunar satellite.

of 50 nautical miles and an inclination to the lunar equator of 4°. The satellite's displacement from its nominal or unperturbed position is due primarily to lunar oblateness. The earth's attraction is 1/85 that of the lunar oblateness contribution on the moon's surface, and the sun's perturbation is 1/160 that of the earth.

The perturbations may be divided into secular, periodic, and short-period variations. The important secular component is a regression of the line of nodes measured relative to inertial space. This effect is shown in figure 29–16 and is about 1° per day. In addition there is a periodic term having an amplitude less than 1° and a period of 1 month. These motions must be considered in addition to the 13° per day rotation of the moon in order to obtain the total relative motion between the moon and the satellite in orbit. These components are known and can be taken into account in the plan of rendezvous guidance.

The short-period displacements of the satellite from its nominal position in circular orbit are shown in the lower portion of figure 29–16. Components are shown for the displacement of the satellite normally, radially, and tangentially from the nominal circular orbit for a period of 6 hours. These displacements are less than a mile in extent and have a period equal to the orbital period of the satellite. These displacements are comparable in size to errors expected in orbit determination, uncertainties in lunar topography, and standard deviations in launch parameters. Because of this fact these displacements can be handled along with other guidance errors.

FACILITIES AND SIMULATORS FOR RENDEZVOUS STUDIES

Many of the guidance and control problems of rendezvous and other phases of the lunar mission are best attacked by analog simulation of the phase of the mission under investigation. This is particularly true when the efficiency of human pilots for accomplishing these phases is being evaluated. Much work has already been done and reported in the area of guidance-and-control simulation, but specific tasks still remain. The following sections give an idea of some of the simulation equipment that is being developed to do this work.

Docking Simulator

One facility that has been used to determine the feasibility of manually controlled orbital assembly and docking operations is the docking simulator shown in figure 29–17. In this simulator two circular light images are projected to represent two close objects in space as seen by a pilot from a third nearby vehicle. These images grow in size and move in relation to the pilot according to control inputs to the pilot's spacecraft and to one of the projected images. An analog computer is employed to solve the equations of relative motion in translation of the two projected images and the pilot's space-

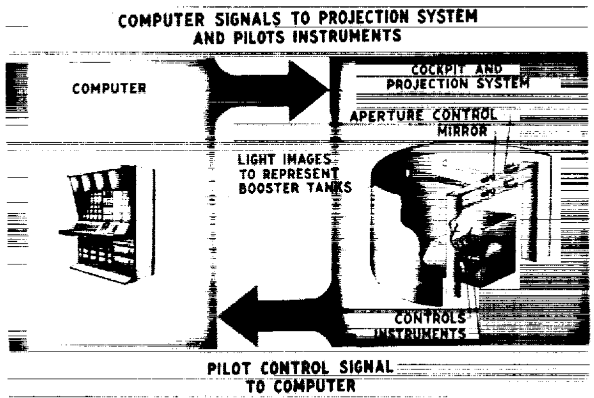

FIGURE 29–17.—Docking simulator.

craft. This simulation thus provides for the study of the assembly of two objects in space from a spacecraft a short distance away or of the ability of a pilot to dock his own spacecraft with another vehicle. An example of the precision with which a pilot is able to control the docking of his spacecraft with one of the projected images by visual cues alone is shown in figure 29–18. These results were obtained in docking operations starting at a distance of 40 feet and for various levels of thrust capability of the pilot's spacecraft from 0.1 to 1.0 ft/sec^2. A perfect dock is obtained when both the lateral displacement y and vertical displacement z are zero at contact. It can be seen that the error in placement at the completion of docking was never more than 1 inch. The velocity at contact was about 0.1 ft/sec, and about 5 minutes

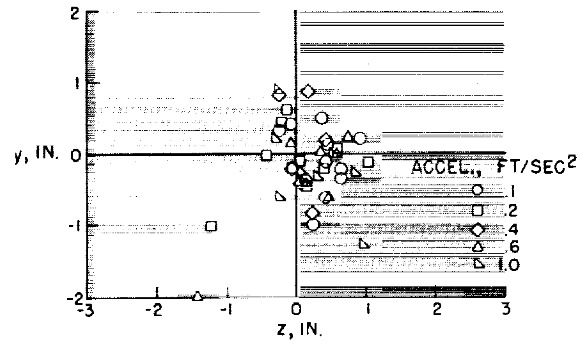

FIGURE 29–18.—Visual rendezous with translation only. Final velocity ≤ 0.1 foot per second; vehicles are attitude stabilized.

was required for the completion of docking. It is mentioned again, as a reminder, that this good performance was obtained by visual cues only.

Gemini Rendezvous Simulator

Another facility that is going into operation shortly is the Gemini rendezvous simulator (fig. 29–19). This equipment is to be used to simulate the manually controlled docking of the Gemini and Agena vehicles. A closed-circuit television system and an analog computer are employed. In this system a small-scale model having three angular degrees of freedom is mounted in front of a television camera. The model translates along the camera axis and rotates in response to the pilot's control inputs and the analog computer. The image of the target is transmitted by the TV system to a two-axis mirror above the Gemini pilot's head and is projected on the inside surface of a 20-foot-diameter spherical screen. Through the added action of this mirror system, all six degrees of

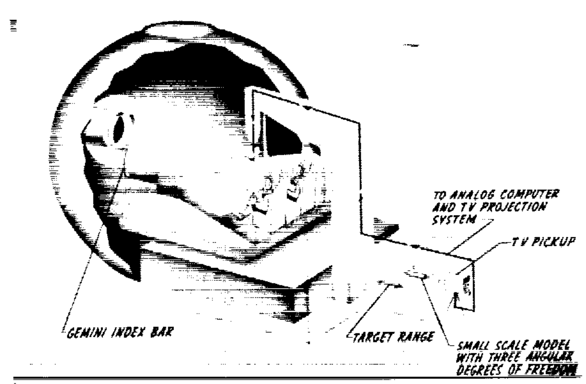

FIGURE 29–19.—Gemini rendezvous simulator.

freedom are simulated. The pilot and crewman are seated in a full-scale wooden mockup of the Gemini vehicle. A moving star field responsive to the Gemini vehicle's angular rates gives an impression of angular motion.

The docking simulation will be initiated with the Agena about 1,000 feet from the Gemini and continued until theoretical contact. This equipment will be used to study the effect of control mode (on-off or proportional controls), thrust levels, system failures, and lighting conditions on the ability of the pilot to perform the Gemini docking operation.

Lunar Letdown Simulator

To study the important and relatively unknown area of descent to the lunar surface, a lunar letdown simulator is being developed. The primary components of the lunar letdown simulator are shown schematically in figure 29–20. This facility uses a closed-circuit TV system in conjunction with four models of the moon and an analog computer to enable a fixed-base simulation of descent from lunar orbit to be made with realistic visual cues. During descent the pilot sees the moon and stars in proper size and relationship to himself through four windows in his spacecraft. One window faces forward, one down, and one to each side. The TV cameras are moved in relation to the models as commanded by the pilot's control inputs and the analog computer, and by this action the views required for the four windows of the spacecraft are obtained. The TV cameras employed are four cameras in one to provide the four views required. The four models employed are at different scales to provide the proper definition of the lunar surface for different altitude ranges, and as the limits of one model are reached automatic switching to the next model will be made to provide for continuous simulated operation. This facility is capable of operating from an altitude of 200 miles down to an altitude of 200 feet. It will be used to study piloting and navigational techniques for manned lunar-landing and lunar-rendezvous operations.

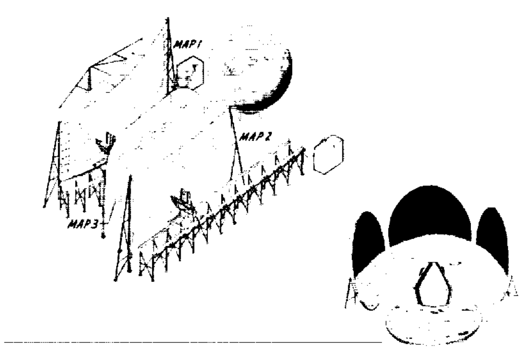

FIGURE 29–20.—Lunar letdown simulator.

Docking Facility

As a further means of studying the rendezvous docking problem, there is under construction a docking facility shown schematically in figure 29–21. This facility employs a spacecraft mockup mounted on a cable and gimbal servo system, an analog computer, and a target vehicle. This facility enables simulation of the docking operation from a distance of 200 feet to actual contact. The servo system moves the spacecraft in response to control signals from the pilot in accordance with the differential equations solved by the analog computer. Six degrees of freedom are simulated in this facility. It will be used to study piloting techniques in docking and in controlling a spacecraft when hovering, during the abort of a landing, or during take-off.

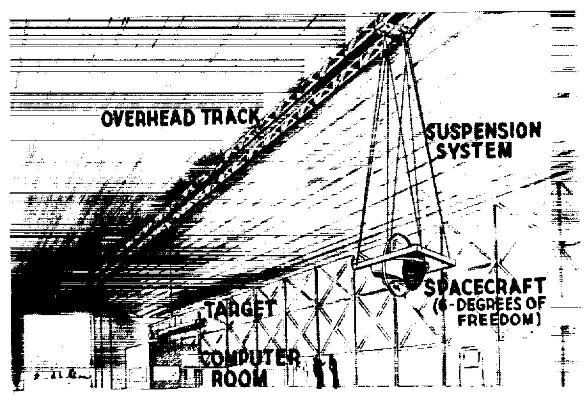

FIGURE 29–21.—Schematic of rendezvous docking simulator.

Lunar-Landing Facility

One of the most critical events envisioned in the lunar-landing mission is the actual landing maneuver. To study this problem a lunar-landing facility is being constructed. This facility consists of a lunar-landing vehicle suspended in gimbals by a bridge and cable system which supports under servo control five-sixths of the weight. (See fig. 29–22.) The spacecraft is free to move in six degrees of freedom under the action of its rocket power and in response to control inputs by the pilot of the spacecraft. The bridge crane system is controlled by a servo so that the suspending cables are always vertical and hence do not restrict the motion of the spacecraft. Longitudinal, vertical, and lateral travels of about 400, 200, and 50 feet, respectively, are provided. The spacecraft will be provided with initial velocities appropriate to the lunar-landing problem by use of a catapult gear. This equipment will be used for studying piloting techniques for lunar landing and problems of visibility, landing abort, and lunar

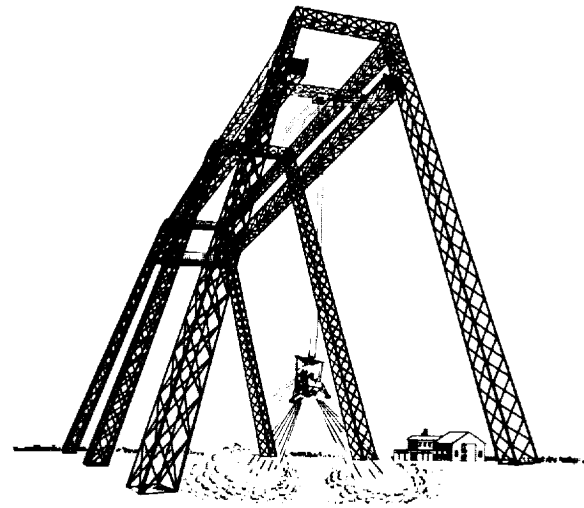

FIGURE 29–22.—Lunar-landing research facility.

hovering and take-off. It is also intended as a further means for studying the rendezvous docking problem.

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N63-11113

30. Atmosphere Entry Guidance and Control

By Rodney C. Wingrove

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INTRODUCTION

Current and future space flight missions require a guidance and control system to regulate the aerodynamic forces during atmosphere entry such that the design limits on acceleration and heating are not exceeded and that the space vehicle at a predetermined destination.

There have been a number of studies of entry guidance problems. This paper will present a survey of this subject taken from the extensive list of published work in this field. The discussions of closed-loop control are taken from references 1 through 36. References 1 through 26 consider primarily automatic control of the vehicle and references 27 through 36 cover principally pilot control. Background material on general problems in entry is taken from references 37 through 57.

This paper will first briefly outline the dynamics in entry and discuss the relationship of the control system and control feedback measurements to these dynamics. The various guidance and control methods then will be considered in some detail. Finally, the capabilities of these systems for entries from circular and supercircular velocities will be covered.

DYNAMICS IN ENTRY MOTION

Before describing the various guidance methods, it is important to explain entry trajectory dynamics. These dynamics are basic to each of the guidance methods and, as in the analysis of any control system, an understanding of the controlled variables is necessary to understanding the over-all control problem.

Dynamics of Constant-Trim Lifting Vehicles

Typical dynamics in atmosphere entry of a constant-trim (constant L/D) vehicle entering the atmosphere are illustrated in figure 30–1.

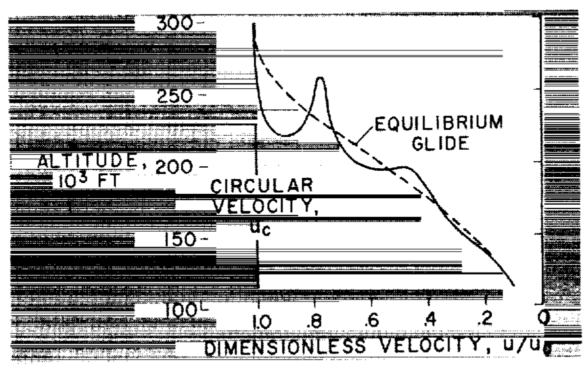

FIGURE 30–1.—Dynamics of constant-trim lifting trajectory.

The vehicle is shown entering at 300,000 feet where the sensible atmosphere begins and also near local circular satellite velocity (u_c). The vehicle is shown to decelerate from these entrance conditions about an equilibrium glide path. An equilibrium glide path is essentially that one path where the aerodynamic lift force just balances the centrifugal and gravity force along the trajectory. The basic vehicle dynamics can be observed from the comparison with this glide path of a trajectory for which the vehicle is not initially in equilibrium (fig. 30–1). It can be seen that the dynamics are stable; that is, the motions are oscillatory and there is a small amount of damping. Many studies (e.g., refs. 24, 25, 37, 38, 39) have analyzed these dynamics. From reference 25 it has been shown that the oscillatory dynamics as a function of the dimensionless velocity, u/u_c, at local points along the trajectory can be approximated by:

$$\text{Damping, } 2\zeta\omega_n \approx \frac{u_c}{u}, \left(\frac{\text{radians}}{\text{unit of } u/u_c}\right) \quad (1)$$

$$\text{Frequency, } \omega_n^2 \approx \beta r \frac{1-(u/u_c)^2}{(D/W)^2}, \left(\frac{\text{radians}}{\text{unit of } u/u_c}\right)^2 \quad (2)$$

The frequency contains the term $1-(u/u_c)^2$ so that the inherent dynamics are statically stable below circular velocity $(u/u_c<1)$, and unstable above circular velocity $(u/u_c>1)$. The light damping for the motion in figure 30–1 is proportional to u_c/u so that the damping increases as the velocity decreases.

The uncontrolled short-period motions of reentry vehicles have been analyzed in references 37–42. The work in reference 39 illustrated that for practical entry configurations these short-period oscillations do not significantly couple with the long-period trajectory dynamics. Most preliminary guidance studies, such as those discussed in the following sections, consider primarily the long-period motions.

Relationship of Control and Dynamics

The effect of the lift and drag forces upon the reentry trajectory dynamics is considered next. The relationship of the control of these forces to the measured state variable in entry is shown in figure 30–2.

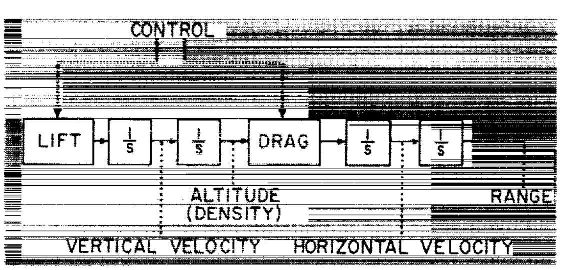

FIGURE 30–2.—Relationship of control and dynamics.

The lift force is essentially in the vertical direction. The vertical force affects the rate of change of vertical velocity. The integration $(1/s)$ of the vertical velocity gives the variation in altitude (or, what is more important, variation in density). This change in density affects the drag force and thus affects the rate of change of horizontal velocity. An integration of the horizontal velocity gives variations in the range along the path.

In order to control range, the rate at which horizontal velocity is changing must be controlled; thus the drag must be controlled. Drag can be regulated principally by either changes in the configuration (i.e., trim changes or drag brake) or, what is more important, changes in density. For instance, if at any time in the trajectory the range must be extended, the lift force is increased in the vertical direction to raise the vehicle into less dense atmosphere, thus reducing the rate of change of horizontal velocity and extending the range.

From this diagram important features in reentry control can be noted and will be referred to in following discussions. The control of range by lift constitutes a fourth-order system (the product of four integrations, $1/s$). It can be reasoned that this fourth-order system, like any other classical fourth-order system, needs four feedback quantities to shape the desired response. In entry the first-order feedback quantity can be the measurement of vertical velocity or can be those measurements that reflect the changing density (i.e., air pressure rate, acceleration rate, temperature rate). For the second-order feedback of altitude variations, those measurements that reflect air density may also be used and, in fact, are desirable because it is the actual density (not the altitude itself) which affects the aerodynamic forces. The third-order feedback quantity can be some measurement of horizontal velocity, and the fourth-order feedback quantity is a measurement of the range-to-go to the destination. Table 30–I is included to illustrate the relationship of possible measurements for control in entry.

TABLE 30-1.—*Relationship of Measurements to Entry Dynamics*

Dynamics / Measuring devices	$\frac{1}{s}$	$\frac{1}{s}$	$\frac{1}{s}$	$\frac{1}{s}$
inertial unit or tracking	vertical velocity	altitude	horizontal velocity	range-to-go
accelerometer	drag acceleration rate	drag acceleration	integration of drag acceleration	2nd integration of drag acceleration
temperature sensor	air or skin temperature rate	air or skin temperature	--------------	--------------
pressure sensor	air pressure rate	air pressure	--------------	--------------

Control Boundaries

It has been shown that the trajectory will be varied to control the range, but the guidance system must also keep these variations within the operating limits. Figure 30–3 illustrates the typical boundaries within which trajectory must be maintained.

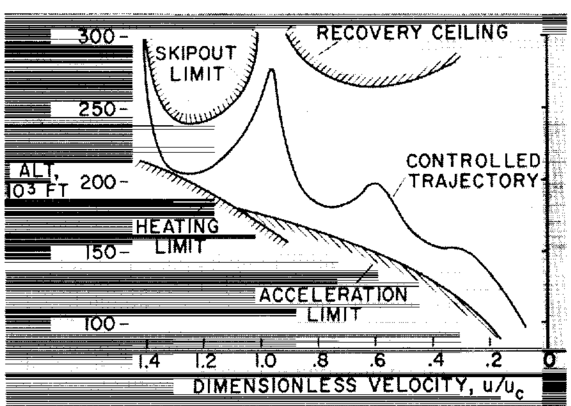

FIGURE 30–3.—Typical operation boundaries.

If a vehicle is at supercircular speed and concurrently reaches too high an altitude at too high a velocity, even though holding full negative lift, the vehicle will skip out of the atmosphere. Also, if the altitude is too high and the velocity too low, the vehicle may not be able to check its descent before passing through the lower boundary. Critical areas shown are those of heating and acceleration limits. A vehicle cannot enter too deeply into the atmosphere because overheating will occur or accelerration limits will be exceeded.

The following discussion will cover various methods that can be used to regulate the aerodynamic forces so that these operating boundaries are not exceeded and the vehicle reaches a desired destination.

DISCUSSION OF GUIDANCE METHODS

This section will outline in detail the various guidance methods. The literature has indicated that each of these methods can yield satisfactory control. The differences to be pointed out in these systems are primarily the relative advantages or disadvantages in the ability to handle off-design entrance conditions; the on-board computer programming requirements; the flexibility to maintain trajectories for minimum heating or minimum acceleration; and the inherent information the guidance equations give a pilot for anticipation and over-all decisions.

The guidance methods examined are presented under two general classifications: guidance using predicted capabilities and guidance using a nominal trajectory. These are further broken down into subgroups. This is not to imply that all guidance systems fall under only one of the categories. These groupings are only made for convenience in the following discussions.

Guidance Using Predicted Capabilities

This general group contains those methods which predict possible future trajectories and do not use a stored nominal trajectory. Figure 30–4 illustrates this method. The vehicle dur-

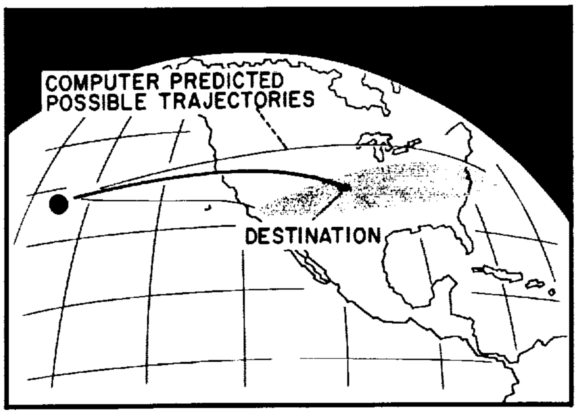

FIGURE 30–4.—Guidance using predicted capabilities.

ing entry has the choice of a number of paths within its maneuvering capability. The guidance system predicts the path by which the vehicle will reach the desired destination without violating the heating and acceleration limits. The discussion of this general method will consider the prediction of possible trajectories by either "fast time" solution or approximate "closed form" solutions of the equations of motion.

Fast time prediction.—The use of fast time prediction has been studied for automatic control in references 9, 20, 21, and 24, and for pilot control in reference 31. The basic concept of these systems is that the differential equations of motion are solved by a "fast" computation in the airborne computer to determine possible future trajectories; repetitive solutions are made from the continuously measured state variables. Typical information needed to make a prediction of the path in the plane of the trajectory is:

1. Four measured state variables (i.e., v, h, u, x).
2. Two vehicle parameters (i.e., L/D, W/C_DS).

Various combinations of these quantities can be used in the prediction. For instance, in the work of references 20, 21, and 31, the altitude and W/C_DS quantities are replaced by the drag acceleration measurement.

The solution of the future trajectory motion can predict maximum excursions of the state variables along the trajectory as well as predicting the vehicle range capability. The computation of heating loads, acceleration loads, maximum skip altitudes and other important constraints can be incorporated into the prediction so that a near optimum trajectory can be followed.

With automatic control, the desired trajectory can be found by iteration. For instance, if in the first computation of the motion equations the desired destination is not achieved, then an error signal is interjected in the next solution. These computations continue until a trajectory is found that will reach the destination. Considerations of the heating and acceleration constraints can be automatically built into the trajectory selection.

With pilot control, the fast-time prediction can be used to display the maneuver capability with respect to possible destinations permitting the pilot to decide upon the proper control actions. From reference 31 it was demonstrated that the simultaneous solution of three trajectories (maximum downrange, minimum downrange, and maximum crossrange) could be used to give the maneuver capability. This repetitive prediction also gives future heating, acceleration, and altitude excursions which may be displayed to the pilot.

The main advantage of the fast-time prediction method is the ability to handle any possible flight condition. This ability to predict range, deceleration, heating, etc., makes it an almost universal control method. The principal disadvantage of the system is the on-board computer requirements for the fast-time computation. Studies to date indicate that the repetitive prediction must be made every few seconds for those entry trajectories where conditions are changing rapidly. In smooth gliding trajectories, computation times on the order of tens of seconds may be permissible.

Closed-form prediction.—For the on-board prediction of trajectories the use of closed-form solutions has also been considered. Closed-form prediction systems differ from fast-time prediction systems in that instead of integrating the equations of motion, they use an approximate explicit solution.

Numerous studies have been made to develop analytical descriptions of entry trajectories (e.g., refs. 43–52). Trajectories that lend themselves to closed-form solutions and have been considered for guidance are, typically, constant altitude paths, constant deceleration paths, ballistic paths, and equilibrium glide paths.

Figure 30–5 illustrates a typical guidance system that uses closed-form solutions for entry from supercircular velocity (ref. 16). The trajectory is divided into three phases of control wherein closed-form solutions can be made. During the flight along the superorbital guide path, a solution is obtained for the range from the measured conditions achievable with a constant altitude path followed by equilibrium glide. The vehicle flies up the superorbital path until this achievable range corresponds to the range-to-go. At that time, the vehicle is

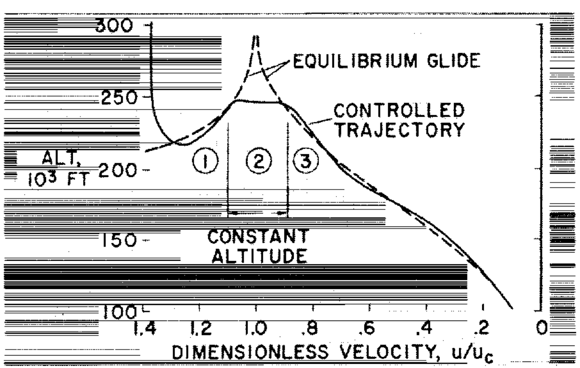

FIGURE 30–5.—Closed-form prediction method.

controlled onto the constant altitude path. The prediction is continued and used along the constant altitude and following equilibrium path, thus providing information needed for control throughout the flight.

Guidance using simple closed-form solutions does not have the flexibility to handle markedly off-design conditions because most closed form solutions do not take account of all state variables needed in the solution of possible trajectories. Also this method has the disadvantage that only those design trajectories that can be analytically described can be used. There is not the flexibility of using any desired trajectory profile such as was the case for the fast-time prediction or will be the case in the following discussion of control using nominal trajectories.

Guidance Using Nominal Trajectory

Guidance about a nominal trajectory is illustrated in figure 30–6. In this general method

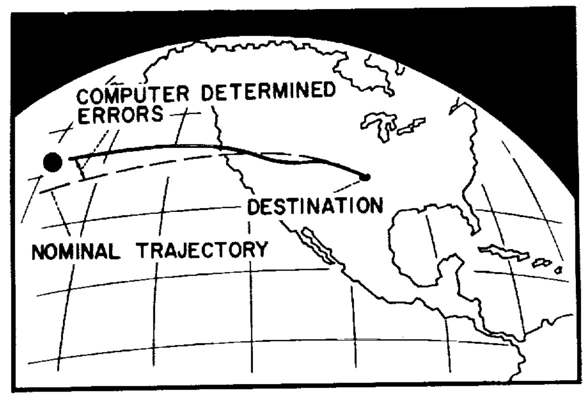

FIGURE 30–6.—Guidance using nominal trajectory.

the state variables along the nominal path are precomputed and stored on board. The variations in the measured variables from the stored values are used in guidance either to control onto the nominal (path controller) or to establish a new trajectory to reach the destination (terminal controller).

For this guidance scheme a trajectory with the most desirable nominal path must be selected. For guidance from subsatellite velocities a constant L/D trajectory close to the center of the vehicle L/D capabilities is generally selected. In most cases, particularly above circular velocity, the problems of acceleration and heating loads, guidance sensitivity, and range capabilities make it necessary to use other trajectories. Various precomputed trajectories are available for analysis in selecting the most desirable flight path; however, the most desirable selections are most probably made by optimization procedures. The steepest ascent computation procedure has been extremely useful in selecting the nominal trajectory. This method has been employed to determine trajectories for minimum acceleration and heating loads (ref. 53) and for optimum lateral turns (ref. 54).

In describing a path through the atmosphere the state variables and, in certain cases, feedback control gains are stored as functions of a given independent variable along the path. Studies indicate that the choice of the independent variable is extremely important, for the guidance system may control through conditions highly divergent from the nominal path. The independent variable may be the obvious quantity, time, or it may consist of one or more combinations of state variables. Those studies using combinations of state variables generally indicate more capability for guidance about a given nominal trajectory than those studies using time. The question of the best independent variable to use in entry guidance is unanswered, though, at the present time.

Path control using fixed feedback gains.—Many studies (refs. 2, 3, 4, 8, 14, 17, 25) have considered the use of constant feedback gains to control onto the nominal trajectory. The guidance law for this type of system with constant feedback gains, K, is:

$$\text{Control } \frac{L}{D}=\left(\frac{L}{D}\right)_{\text{nom}}+K_1\delta v+K_2\delta h+K_3\delta u+K_4\delta x \qquad (3)$$

where $(L/D)_{\text{nom}}$ is the value along the nominal trajectory and the δ quantities represent the deviations of the state variables from the stored values along the trajectory. In applications of this law where the controlled trajectory does not deviate greatly from the nominal, one or more of the $K_1\delta v$, $K_2\delta h$ or $K_3\delta u$ terms may possibly be unused because, as shown in figure 30–1, the basic dynamics in entry do possess some stability. All four state variables should be used in applying this guidance law to conditions where the path does deviate greatly from the nominal or in considering those portions of the trajectory where the guidance situation is extremely sensitive, as in control at supercircular velocities.

Numerous system concepts have proposed using simplified versions of the constant feedback control method. Using one of the state variables (i.e., velocity, range-to-go, altitude) rather than time as the independent variable will simplify the guidance function. Other concepts propose the use of an electrical compensation network (i.e., lead or lag) to replace the number of measurements of state variables. For instance, a simple method for control can be of the form:

$$\text{Control } \frac{L}{D}=\left(\frac{L}{D}\right)_{\text{nom}}+\left(K_1 s+K_2+\frac{K_3}{s}+\frac{K_4}{s^2}\right)\delta a \qquad (4)$$

where δa is the deviation in the measured drag acceleration from a programmed acceleration time history. To give guidance precision to such a scheme, though, the guidance program must be updated periodically from other sources.

The fixed gain systems will work in many reentry applications. The use of varying gains, which will be discussed next, allows more freedom to shape the desired response and give a more optimum control along the trajectory.

Terminal control using influence coefficients.—One useful method of examining control about a nominal trajectory is to employ influence coefficients. These functions are solved from the set of differential equations adjoint to the linearized perturbations of the equations of motion about a nominal path. The methods of adjoint functions have been applied to entry guidance in references 7, 10, 11, 13, and 15, and only the final results of these studies will be given here. The expression for $\delta \mathbf{x}$ at final time is:

$$\delta x|_{t_f}=\lambda_1\delta v+\lambda_2\delta h+\lambda_3\delta u+\delta x|_{t_i}+\int_{t_i}^{t_f}\lambda_b b\,dt \qquad (5)$$

In this equation λ represents the time varying influence coefficients that relate the final response at t_f to the unit initial conditions at t_i. The symbol b is the forcing function (i.e., lift or drag) of the original perturbation equation.

One guidance concept wherein these influence coefficients are used to predict the final conditions is the constant-trim terminal control. This guidance method uses a control input that the influence coefficients indicate can be held constant to reach a selected destination. In closed-loop control this value will change slightly with the continuous measurements that update the predictions. This control corresponds to a step input in the forcing function b of equation (5).

Such a constant-trim guidance method is only good for the linear region near the nominal trajectory. To handle conditions far removed from the nominal, overcontrol instead of constant-trim control is needed to assure the destination remains within the vehicle maneuver capability. Also, because the prediction may be in error as the result of measurement errors, atmosphere density variations, aerodynamic trim variations, and other uncertainties, overcontrol is needed.

Overcontrol to permit guidance from conditions far removed from the nominal trajectory can be accomplished with bang-bang terminal controllers. These controllers use maximum effort to force the path onto a new trajectory to reach the destination. From equation (5) this corresponds to maximum control effort until the final value is driven to zero; that is:

$$\delta x|_{t_f}=0=\lambda_1\delta v+\lambda_2\delta h+\lambda_3\delta u+\delta x|_{t_i} \qquad (6)$$

Some dead band would actually be used to prevent a continuous limit cycle operation. When the final value predicted falls within this dead band then only nominal control effort is required while the vehicle travels along a new trajectory. Guidance methods of this type have been used in reentry studies to control from circular velocity both lift-modulated (ref. 7) and drag-modulated (ref. 18) vehicles and also to control a combination lift and drag varying configuration in pull-out maneuvers at supercircular velocities (ref. 15). The general form of the linear prediction control law can be expressed as:

$$\text{Control } \frac{L}{D}=\left(\frac{L}{D}\right)_{\text{nom}} + K\,\frac{\partial(L/D)}{\partial x}\,(\lambda_1\delta v+\lambda_2\delta h+\lambda_3\delta u+\delta x) \qquad (7)$$

where for constant-trim control $K=1$ and for bang-bang control $K=\infty$. Any value of gain between $K=1$ and $K=\infty$ can conceivably be used to obtain the needed overcontrol. In determining the best value of overall gains to use in the control system of this general type, the optimization procedures as described in the next section should be considered.

Optimized feedback gains.—With optimized feedback gains the guidance will either control onto the nominal path or establish a new path to the destination. In optimizing the set of time varying feedback control gains there is the choice of a performance index which the particular optimization procedure will attempt to minimize. With this design choice, what may be optimum in one case may not be optimum in another. The important consideration is that the system satisfies the requirements and constraints established. The optimization procedures provide the technique for carrying out this control system analysis in a direct and systematic manner. This paper will outline these optimization studies which have dealt directly with reentry guidance and will not attempt to discuss all the optimization procedures which have been developed.

In reference 10 the lambda matrix control scheme is applied to the guidance of a low L/D vehicle entering the atmosphere at supercircular velocity. Lamda matrix control considers the final value (terminal control) and minimizes the mean square control deviation. The control law for reentry guidance is of the form:

$$\text{Control } \frac{L}{D}=\left(\frac{L}{D}\right)_{\text{nom}}+\Lambda_1\delta v+\Lambda_2\delta h+\Lambda_3\delta u+\Lambda_4\delta x \qquad (8)$$

where Λ represents the time varying gains determined by the optimization procedure. The time varying gains are precomputed and depend upon the design nominal path.

In reference 26 the neighboring optimum control scheme is used in guiding a low L/D vehicle from supercircular velocity. This system uses time varying gains that optimize one of the terminal quantities while satisfying other terminal constraints.

In reference 24 the parametric expansion method is applied to the control of a medium L/D vehicle reentering at circular velocity. The performance index considers the integral squared error of the variations in dynamic pressure, range-to-go, vertical velocity, temperature, and the angle-of-attack control effort.

Additional considerations for the crossrange requirements are the integral squared error of the crossrange-to-go and bank-angle control effort. Weighting functions are considered with the various terms to determine the time-varying gains.

These optimization studies for reentry control have all used time as the independent variable. Consideration has not yet been given for the use of state variables (i.e., velocity, range-to-go, etc.) in the independent variable.

In order to summarize the discussion of guidance methods using a nominal trajectory, figure 30–7 illustrates the various schemes. The fixed

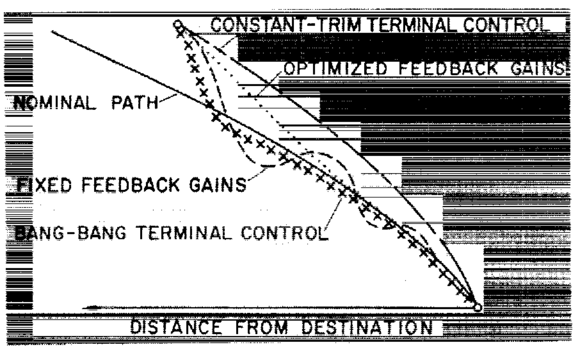

FIGURE 30–7.—Various guidance methods using a nominal trajectory.

feedback gain method of control produces a trajectory which oscillates about the nominal trajectory as shown on the figure. The constant-trim terminal controller method uses an essentially constant control input to establish a new trajectory to the destination. The bang-bang terminal controller method uses maximum control input until the nominal control can be employed to establish a new trajectory to the destination. The optimized feedback gain method uses time-varying feedback gains that have been optimized either as a terminal controller or as a path controller about the nominal trajectory.

The fixed feedback gain method provides good control primarily below circular velocity. The other varying feedback gain methods (terminal control and optimized feedback gains) generally provide better over-all capability. All of these methods are relatively simple guidance systems which require a minimum on-board computer. A disadvantage of using these methods compared with using the predicted capabilities methods is the minimum amount of information available for a pilot display. This is discussed further in the following section.

Pilot Participation in Guidance

The information that the pilot receives from the guidance computer comes from displays of the type shown in figure 30–8 depending upon

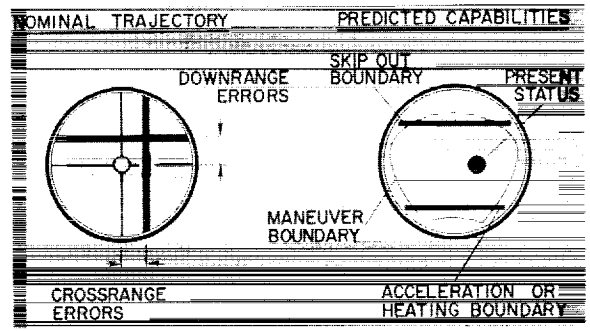

FIGURE 30–8.—Typical guidance displays.

the particular guidance equations. For the guidance method which uses a nominal trajectory the information shown in the figure is available; this is similar to the command needles in current aircraft. For the guidance method which uses predicted capabilities additional information may be available to give the remaining maneuver capability and to anticipate possible skip out, heating and acceleration problems.

In manned vehicles the capabilities of the pilot may be considered as another loop around the basic guidance system (refs. 27–36). A system's reliability is increased when the pilot monitors radio signals, navigation equations, acceleration sensors, temperature sensors, and other data essential to entry guidance. When necessary, the pilot can override incorrect or questionable signals, provide the control responses, and make decisions on the over-all mission profile.

Typical Vehicle Control Systems

The regulation of the lift and drag forces has been considered in the foregoing general discussions. In vehicle control the actual lift or drag usually need not be measured but, instead, the measured flap deflection or angle of attack (α) and roll angle (φ) which correspond to variations in the lift or drag are used in the control loop. The following two control schemes illustrate typical proposed vehicle systems.

For the control of the Dyna Soar boost glider at subcircular velocities, it has been proposed (ref. 14) to control downrange in the following manner

$$\text{control } \alpha\Big]_{\alpha_{\min}}^{\alpha_{\max}}=\alpha_{\text{nom}}+K_1 v+K_4\delta V\Big|_x \qquad (9)$$

The nominal trajectory consists of a stored program of total velocity, V, versus range, and a filtered vertical velocity term v is used for damping. Crossrange is controlled by;

$$\text{Control } \varphi=K\psi \qquad (10)$$

where ψ is the angle between the instantaneous heading of the velocity vector and the heading of the great circle line between the present position and the landing site.

For illustrative purposes, we can consider the same general method of control about a nominal trajectory for a low L/D, constant-trim lifting body that is typical of those considered for entries at super-circular velocity in the return lunar mission. For a constant-trim lifting body, the roll angle can be used to control the lift vector in the vertical plane in the following manner (ref. 17).

$$\text{Control } |\varphi|\Big]_{180^\circ}^{0^\circ}=\varphi_{\text{nom}}+K_1\delta v+K_2\delta a+K_4\delta x\Big|_u \qquad (11)$$

The three state variables of the nominal trajectory are stored as a function of the horizontal velocity. The gains K can conceivably be constant but, for the best control system, they should be variables determined by optimization design procedures. This command equation determines the magnitude of the roll angle. The sign of the roll angle is determined by the heading to the landing site.

GUIDANCE CAPABILITIES IN ENTRY

Attainable Ground Area

The previous discussions have outlined possible guidance methods. The ground area attainable in the studies of these systems to date is shown in Figure 30–9. As shown for reentry

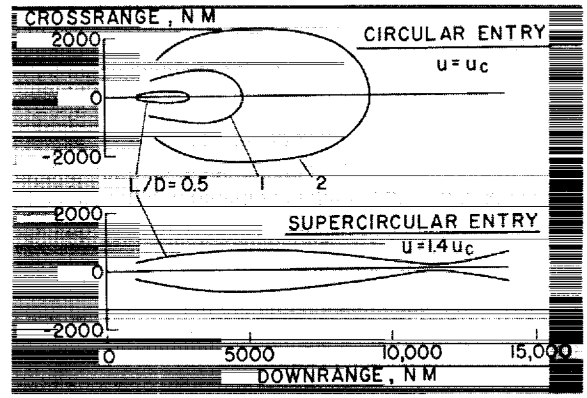

FIGURE 30–9.—Attainable ground area.

from circular orbit the available lift-to-drag ratio of the vehicle is extremely important in determining the attainable ground area. For those vehicles with an L/D below 0.5, such as the proposed lifting Mercury configuration, there is about 200 miles crossrange and 2,000 miles downrange capability. For the higher L/D vehicles such as the Dyna Soar with an L/D between 1 and 2, there is much more ground area available for maneuvering from orbit.

Most guidance studies of closed-loop control for entry at supercircular velocity have considered a low L/D vehicle. As shown in this figure, a vehicle with an $L/D=0.5$ at supercircular velocity has about 600 miles of maximum crossrange capability and it is interesting to note that at about 12,000 miles from entry the crossrange is limited. This is because the vehicle is one-half the way around the Earth and the great circle routes from the initial entry converge at this point. Entry at supercircular velocity, such as will be encountered in the return from a lunar mission, presents particular guidance problems which will be discussed next.

Guidance from Supercircular Velocities

To make a successful entry from supercircular velocities the vehicle must be on a trajectory within a safe entry corridor (Refs. 55 and 56). This corridor is shown in Figure 30–10 in relation to the overshoot boundary, where the

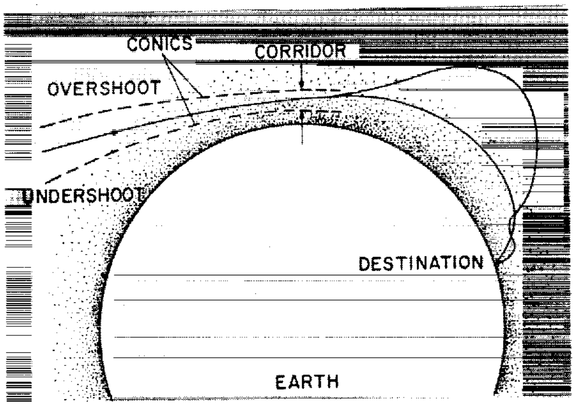

FIGURE 30-10.—Guidance in supercircular entries.

vehicle will just stay within the atmosphere, and the undershoot boundary, where the vehicle will reach the specified acceleration or heating limit. From within this corridor the reentry guidance system can control the vehicle either through the atmosphere or on a skipping maneuver out of the atmosphere to the destination. The choice of the particular design trajectory is a compromise between acceleration and heating loads, guidance sensitivity and the range to go to the target.

The available range control within the operational corridor for a $L/D=0.5$ vehicle is shown in Figure 30-11. The upper boundary of the corridor is determined by the shallowest entrance at which the vehicle can remain within the atmosphere while holding full negative lift. Control near this limit is sensitive because near full negative lift is needed to keep the vehicle within the relatively low density position of the atmosphere and little lift is available for shortening the range. Also near the overshoot

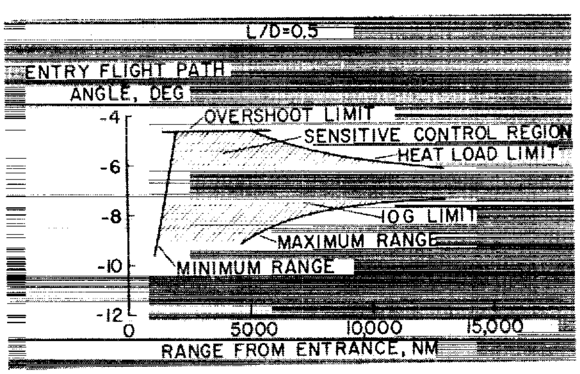

FIGURE 30-11.—Typical operational corridor for supercircular entry.

boundary the vehicle spends a long time within the atmosphere so that the total heat load limit can be exceeded in controlling to the longer ranges. The undershoot limit is determined by the acceleration limit of the vehicle or crew. The value of $10g$, usually used in defining the corridor, has been shown in Reference 57 to be a realistic value within which humans can still perform. With control at the steeper entrance angles the maximum range capability of the vehicle is limited because a large amount of vehicle kinetic energy is lost during the initial steep dive into the atmosphere.

For supercircular entry velocities the range capability from about 1,500 to 12,000 miles has been achieved in guidance studies to date. This range capability has been demonstrated in references 20 and 21 using the fast-time prediction method. Recent studies by Lessing and Coate of NASA Ames Research Center have demonstrated that this capability can also be achieved using control about a single nominal trajectory.

The direct descent entries have been the primary object of the supercircular entry studied to date. There has been some consideration of establishing a circular orbit in the skip-out (refs. 12, 50) and the use of multiple pass braking (ref. 55) but there has been little analysis of the closed-loop control to mechanize these schemes.

CONCLUDING REMARKS

In conclusion, this survey has outlined several guidance and control methods that have been proposed for atmosphere reentry. Each method can be designed to realize the vehicle capabilities near the design entry conditions. It has been found important that the measurement of all four state variables should be represented in the guidance law to insure control in sensitive guidance situations or markedly off-design conditions.

Further research is warranted to determine which measurements should be used to represent the state variables and how to best use these variables in the guidance systems. Also the more stringent mission profile of the advanced space flight projects warrant additional research toward the development of new guidance systems and toward the improvement of the systems outlined in this survey.

NOTATION

a measured acceleration, g units
b forcing function
C_D drag coefficient
D drag force; along the total velocity vector, lb
h altitude; along radial axis, ft
K constant feedback control gain
L lift force; normal to the total velocity vector, lb
r distance from planet center, ft
S surface area upon which force coefficients are based, ft^2
s differential operator notation
t time
u circumferential velocity; normal to radius vector, ft/sec
v vertical velocity; along radial axis, ft/sec
W weight of vehicle, lb
x downrange to destination, along great circle route, nautical miles
α vehicle angle of attack
β atmosphere density decay parameter, ft^{-1}
δ denotes deviation in quantity from that of nominal trajectory
λ influence function
Λ time varying feedback control gain
φ vehicle roll angle
ψ heading angle between the instantaneous great circle route and the great circle route to the target
ζ damping factor
ω_n natural frequency

Subscripts:

f final value
i initial or instantaneous value
nom respect to nominal trajectory

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31. Space Vehicle Attitude Control

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INTRODUCTION

As navigation refers to the problem of controlling the translational motion of a vehicle, so attitude control refers to the problem of controlling the rotation of a vehicle.

The two types of control exercised are: (1) turning the vehicle in a prescribed manner to a prescribed degree; and (2) stabilizing the heading or attitude in a desired direction despite the disturbing action of unwanted torques. Table 31–I shows some of the functions required of vehicles and gives an indication of the accuracy with which they are to be performed. The functions are divided into the maneuvering a vehicle performs in first becoming oriented, and the maneuvering and attitude holding it performs in its normal course of operation. Outside of the operations a vehicle performs in becoming initially alined, the maneuvering accuracy required of a vehicle can be as high as a minute of arc for certain astronomical applications. The requirement reduces to the order of degrees when solar arrays are to be turned toward the Sun. There is a greater accuracy spread in the stabilization required of vehicles: In measuring spectral characteristics of stars a vehicle may be called upon to hold its position to within 0.1 second of arc for over 2 hours; on

Table 31–I.—*Example Attitude Control Functions and Accuracy Requirements*

Initial orientation		Attitude Change		Attitude stabilization	
Operation	Accuracy	Purpose	Accuracy	Attitude	Accuracy
Stop tumbling	0.005 deg/sec	Measurement	1 min	Toward Sun	1 sec–several deg.
Extend arrays		Solar power	Several deg	Toward star	0.1 sec.
Aline toward Sun	¼–1 deg	Thrust vector alinement.	0.3–1 deg	Toward vertical	0.1–several deg.
Roll about sun-line.	0.2 deg/sec			Zero yaw angle	1–several deg.
Acquire star	0.5 deg				
Acquire Earth	Several deg				
Aline experiment package.	Several deg				

the other hand, the vertical orientation or the heading of the vehicle or some of its parts may not have to be controlled closer than to within a few degrees.

An abbreviated list of the missions calling for these operations is given in table 31–II. Important limitations on the designs can be found under the headings: expected "lifetimes" and date of "first launch." The nearness of the times shown by the launch dates listed implies that the control systems are to be designed using available techniques and available or nearly available equipment.

It is not our purpose today to give detailed descriptions of the operations required of space vehicles or of the environmental factors influencing control system design nor to give details of the means of securing a desired control. We will touch upon them as needed, but they are considered more completely in the references (refs. 1, 2, and 3). Rather, this paper intends to point out some of the limitations imposed by the present state of the art upon some current control system designs and to indicate the trend being followed in the removal of these limitations.

TABLE 31–II.—*Missions and Orbits of Selected Satellites*

Vehicle	Mission	Launch	Life (Months)	Attitude	Altitude (Nautical miles)	Inclination (Degrees)	Weight (Pounds)
Syncom	Communications.	1963		Inertial	19, 000	33	73
Tiros	Meteorology	Apr. 1960	3	Inertial	400	48	280
Nimbus	Meteorology	1963	6	Vertical	600	80. 5	650
OSO I [1]	Scientific	Mar. 1962	6	Solar	300	33	450
POGO [2]	Scientific	1963	12	Vertical	200–600	33	950–1, 500
EGO [3]	Scientific	1963	12	Vertical	150–60, 000	33	950–1, 500
OAO [4]	Scientific	1964	12	Celestial	400	33	3, 300

[1] Orbiting Solar Observatory.
[2] Polar Orbiting Geophysical Observatory.
[3] Eccentric Orbiting Geophysical Observatory.
[4] Orbiting Astronomical Observatory.

GENERAL DESCRIPTION OF ATTITUDE CONTROL SYSTEMS

Figure 31–1 summarizes the structure of a vehicle attitude control system. The system is seen to consist of a vehicle, of an evaluation of its performance relative to a reference behavior, and of control. The evaluation portion of the system embraces sensors of vehicle behavior, a

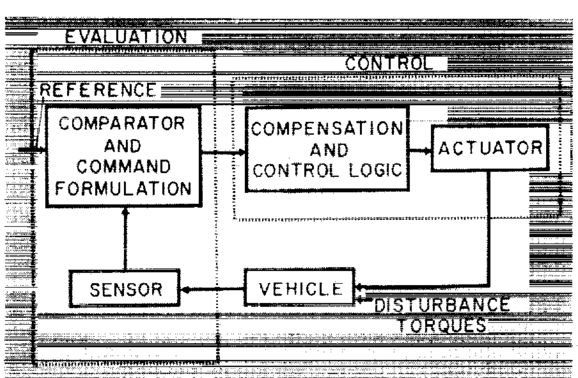

FIGURE 31–1.—Attitude control system elements.

comparison with a reference behavior, and an interpretation of discrepancies in terms that are usable by the vehicle. The control portion of the system operates on this information formulating some law by which to govern the next behavior of the vehicle.

The elements constituting each portion are as varied as the functions to be performed. In the evaluation portion, the sensor might be a solar eye giving a voltage output that varies over a certain range, depending on the direction between it and the Sun. If the device is designed to have no output when it is pointing toward the Sun, then the reference and comparator are built into the device and the output is assumed to form a command signal. Depending on its purpose and design, such a device may give an indication of the direction to the Sun with an accuracy on the order of degrees or seconds of arc.

Another example is furnished by a rate gyro used to provide yaw information in vertically oriented vehicles. If the vehicle maintains its vertical attitude, then it turns at the rate of one revolution per orbit. The gyro is arranged to measure the rate at which the vehicle rolls about its flight direction or a component of the orbital rate if it yaws about the vertical. If the roll rate is kept zero in some fashion, the gyro voltage output is proportional to the yaw angle. In this example again the reference is implied by the zero of the gyro output signal. Nonzero values are used as commands to the vehicle controller. This arrangement, which is likely to be used together with a horizon sensor to furnish an indication of roll, is not very sensitive but can be used to keep the yaw angle to within a degree or so of zero.

A pair of star trackers can be used to provide commands that are sufficient to stabilize a vehicle to less than half a minute of arc. These small telescopes closely track a star independently of the motion of the vehicle. The angles between the telescope axes and the vehicle are measured then compared with stored reference angles. The differences are processed according to the geometry established by the star trackers and the vehicle to indicate the position error of the vehicle. These calculated errors then act as commands for the controller.

Commands undergo a certain amount of processing—which will be considered later—before driving the actuators. The actuators, the muscles by which the desired torque is exerted on the vehicle, are usually of two types: momentum removal and momentum exchange.

Momentum exchange devices are exemplified by reaction wheels which, in being driven, react on the vehicle in a direction opposite to that in which they are speeding up. By their turning they absorb the momentum of the vehicle which is turning in an undesired way. Reaction wheels are advantageous when accuracy of attitude control is desired, when disturbances of the vehicle are largely oscillatory, and when a relatively long lifetime is required.

Momentum removal devices eject matter from the vehicle, the reaction to the ejection torquing the vehicle. A jet of compressed nitrogen exhausted through a nozzle is an example of a momentum removal device. Jets which have a higher torque capability than wheels are used to stop initial tumbling of vehicles. They are used also to dump the momentum stored in reaction wheels which are approaching their speed limits. Since they have the disadvantage of using up fuel whereas reaction wheels can obtain power from the Sun, jets are not generally suitable at present as prime actuators on long-lived missions. They are appropriate for interplanetary trips, however, where the vehicle disturbances are not oscillatory.

Vehicle disturbances are unwanted torques superposed on the vehicle in addition to the torque exerted by the actuator. Examples representative of disturbances for such vehicles as Nimbus or the OAO are shown in Table 31–III. Some torques arise internally from the vehicle's own dynamics or from motion of its parts. Others are due to environment—solar radiation pressure, micrometeoritic bombardment, the Earth's atmosphere, or, as indicated on the table, the gradient of the Earth's gravity field or the geomagnetic field.

The predictable torques among those on this list, particularly those from the gravity gradient or the geomagnetic field, may not always be unwanted. They can be used directly or indirectly together with actuators for vehicle stabilization.

TABLE 31–III.—*Typical Attitude Disturbance Torques*

		Magnitude
Vertically Oriented Vehicles	Torque in roll due to yaw angular velocity	50 dyne cm./deg./hr.
	Torque in roll due to roll angle	200 dyne cm./deg.
	Torque in pitch due to orbit eccentricity	200 dyne cm.
	Torque in pitch or roll due to gravity gradient	500 dyne cm./deg.
Inertially Oriented Vehicles	Torque due to gravity gradient	2,500 dyne cm.
	Torque due to magnetic field	2,500 dyne cm.

PRESENT STATUS OF ATTITUDE CONTROL SYSTEMS

Evaluation

The previous remarks were made to give a general idea of the structure of a space vehicle attitude control system and of some of its functions. The expected lifetimes and the short lead time called for in designing and building satellites, in forcing the designer to use the simple methods and reliable equipment that is either already on hand or to be available very shortly, have necessitated some solutions which fall short of a designer's ideal. Examples of compromises in performance or flexibility due to the realities of present capability can be found in those parts of the control system labeled "evaluation" in figure 31–1. Compromises are made sometimes because of the limitations of available sensors, and at other times to avoid performing complicated calculations on board the vehicle.

Consider the common problem of keeping one axis of a vehicle vertical and another in the orbit plane, that is, with zero yaw angle. For a hypothetical mission calling for great precision in maintaining this attitude, the very first step in design requires that distinctions be made between possible interpretations in these references. After all, the so-called orbit plane is almost always only a useful fiction, and gravitational and geometric verticals need not coincide. Suppose for the example, however, that a geometric vertical is desired and that a vehicle reference axis in the local horizontal plane has zero yaw if it has no angular rate component along it. Then the figure of the Earth and orbit inclination set fundamental limits to the position and rate of the vehicle's vertical attitude. In practice, however, noise from a horizon sensor limits accuracy. The noise might arise from predictable sources, such as the Sun on the horizon, or from less predictable sources, such as cold clouds the existence of which took flight experience to discover, or from the basic limitation imposed by the temperature of the sensor. Whatever the source of noise, means can be devised to circumvent it or to improve the signal accuracy. The sensor can be redesigned, estimators and predictors can be formulated to operate on the signal, or a new type of sensor may be used. If the mission is to be flown in the next six months or so, however, these problems cannot be gone into, and the mission cannot expect a better position accuracy than that of one or two degrees.

Similar remarks apply to the measurement and control of the yaw angle. Difficulties here are associated with uncertainties not only in the threshold of the measurement but also in the operations used to determine that a nonzero signal is caused by yaw. In this case, too, optimal estimations will improve the resulting commands. But the necessary methods are not sufficiently developed to be currently employed. Missions in the near future cannot expect yaw control of higher precision than the order of one degree.

Methods not only of implementing the results of decision and optimal estimation theory but also of computing command equations must be improved before they can be employed on board a vehicle. There are instances among current vehicles that require some operations to be performed on the ground in order to avoid the currently undesirable task of computing variable command equations on board.

The command equations used to evaluate the performance of a vehicle in order to control it are generally determined by the geometry of the situation. Since the equations are often nonlinear, they must normally be simplified and approximated before they can be used. The extent to which the stable range of system operating errors depends on approximations in commands is a fundamental control problem about which there is little knowledge. Much experience, however, has shown that the approximations used in the equations are critically important to the stability of the whole control system. In the case of one vehicle, whose attitude changes are sensed by a number of star trackers, very approximate but simple command equations are mechanized on the vehicle. Then every significant change in commanded attitude is checked by a computation on the ground to determine whether the system will be stable with the vehicle in the new attitude.

The great degree of approximation used in command equations will be reduced as computers designed for space vehicle operation become available. Improvements that will let computers be designed into a system with confidence in their long-term operation and minimum complications in other operations of the vehicle will be of great value in increasing the flexibility and precision of a number of attitude control systems.

Control

Except for the development of such new instruments as horizon and solar sensors, the evaluation portion of space vehicle control systems has not progressed beyond the older technology for aircraft. Greater change has occurred in the control portion of the system. The greatest change here has been an increasing dependence on nonlinear analysis and design and the employment of increasingly sophisticated and effective control logic.

The central design problem in control systems used to be selecting compensation for command signals which were typically position error signals lineraly dependent on the vehicle's behavior. The problem was to select such proportionality factors and judicious amounts of derivative and integral functions of the error that the over-all system behavior became not only stable but in some sense fast. Effort was expended in securing linear performance from each element of the system not so much because linear elements behaved so well as because their behavior could be predicted so well. Although the virtues of relay and other nonlinear systems were recognized by some theorists and experimenters, nonlinear techniques were seldom used or needed.

The control design problem in satellites is one of conserving power rather than of securing speed. Whereas linear systems expend power continuously, pulse systems do not. Use of jet pulses to control the attitude of space vehicles, however, forces the designer to apply nonlinear techniques in the system analysis and mechanization. The position error of the vehicle is no longer continuously forced toward zero, but oscillates in a limit cycle, a naive sketch of which is shown in figure 31-2.

In this figure, position error and error rate—the time derivative being designated by the dot

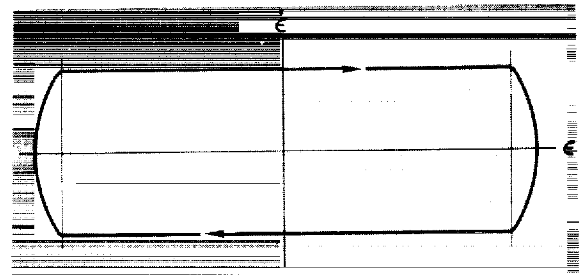

FIGURE 31-2.—Idealized vehicle limit cycle behavior.

above the letter—are used as abscissa and ordinate. The error increases at a constant rate until a certain value when gas is ejected. The gas impels the vehicle to move with a rate of the opposite sign. The error then decreases until it is again of a certain size. Another pulse of gas returns the vehicle to its initial motion.

It is clear from the diagram that to conserve fuel, the time between pulses should be as long as possible. Some logic schemes strive to stretch the oscillation period by modulating pulse magnitudes by noting the time between pulses. Others—for inertially oriented vehicles—rather than estimate the time between various errors, use vehicle rates computed by derivative networks. It is even possible to devise schemes that, in effect, remember and use ambient fields to increase the time between pulses.

Partly the necessity of using nonlinear error signals and partly experience with jet controls have given an impetus to design equivalent logic schemes for use with reaction wheels. Reaction wheels can be built as linear devices that depend linearly on speed. To obtain most momentum storage, however, while keeping a uniform torque capability, nonlinear reaction wheel characteristics which are independent of speed over part of the operating range have been found useful. Since such wheels behave like pulse jets, the same sort of control logic used for jets has been applied to them.

CONCLUDING REMARKS

This paper has presented a brief view of the present state of the art of space vehicle attitude control. The status revealed was biased toward the limitations in current systems in order to point out the trends of future development.

As selective as the paper has been, it ought not be closed without having indicated the background knowledge needed by a person wishing to work in this field.

But a few years ago the tools of the control systems specialist consisted of a knowledge of differential equations, some transform theory, linear network analysis and synthesis, geometry and kinematics, and a knowledge of the properties of transducers. But the field of control theory has been infused with a new viewpoint and a number of new ideas. Now to exploit the theories of optimal estimation, optimal control, adaptive control, and dynamic programming, to amplify the ideas of pattern recognition, or to investigate the topological approaches to stability theory, much mathematics new to the control specialist is required. Knowledge of information theory, game theory, statistical decision theory, of linear spaces and nonlinear differential equations, of integral equations and variational calculus is becoming increasingly important.

An increase in breadth of mathematical background seems to be a requirement common to all technical endeavors these days. Yet mathematical capability is not nearly all that is needed today in the design of control systems on the basis of modern theory. Adaptive systems, optimal control systems of all sorts, even the more traditional approaches when applied to the more complex situations, all call for digital computers on board the spacecraft. The design of these devices is very promising right now. But their use may well prove to be a crutch to systems and components designers. To the extent that we look forward to the growth in numbers and capabilities of control theorists, we also look forward to finding gifted experimenters with the ingenuity to devise equipment that will accomplish passively some of those things that we tend now to rely on computers to accomplish.

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N63-11115

32. Guidance and Control Components Research

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INTRODUCTION

This paper will examine some research and development efforts on advanced spacecraft guidance and control components and systems. Some of the difficulties encountered in system design due to neglecting characteristics of real hardware are noted as well as the constraints imposed on component development when tied to space programs. The unique position of the universities to make contributions to this field are also discussed.

EQUIPMENT RESEARCH AND DEVELOPMENT FOR THE NASA

Research on navigation, guidance, and control equipment used on NASA spacecraft is carried on by many organizations. Advanced spacecraft equipment is based on the results of the research by the following:

1. Research at NASA research centers.
2. Research at NASA space flight centers.
3. Industry-sponsored research and development.
4. NASA contracts and grants.
5. Major space vehicle projects.
6. Department of Defense, universities, etc.

Each of these organizations operates with somewhat different immediate objectives and constraints; nevertheless, they all have the same ultimate goal of improving the state of the art. The in-house programs of the NASA Research Centers, such as Ames and Langley, are concerned primarily with the advanced techniques needed for both present and future space projects. In general, these studies are not directed toward hardware development, particularly flight hardware, but rather are concerned with the study of new concepts. An important by-product of this research effort is the development of in-house groups with sufficient knowledge of the state of the art to intelligently judge contractors' proposals and subsequent development work. The efforts of the NASA space flight centers, such as Goddard and Marshall, are similar to the research centers; however, a much larger proportion of their effort is directed toward solving problems of immediate concern to existing programs.

Industry-sponsored programs are very valuable, particularly in the component area. Programs supported by the NASA Office of Advanced Research and Technology provide a wide base for research and development in all of the space technology fields. Their contract program is particularly useful for exploiting new concepts proposed by industry. The NASA university program, which consists primarily of research grants, also provides an excellent source of research talent.

Major space vehicle projects such as the Orbiting Astronomical Observatory and the Apollo program provide a large portion of the funds spent on equipment research and development. By the nature of these flight projects, the emphasis is on hardware development; however, advanced concepts of these vehicles require a significant effort in applied research.

NASA RESEARCH CENTER PROGRAMS

The magnitude of NASA in-house research is considerably smaller than that accomplished under contract; nevertheless, it constitutes an important segment of the over-all program because of its advanced nature, and also because it provides the continuation of an in-house competence to judge proposals soliciting NASA support.

A typical NASA Research Center program concerned with real hardware systems is represented by the work of the Guidance Systems Components Branch at the Ames Research Center. The activities of this group are representative of an effort to combine theory with the practical in the development of guidance and control components. Figure 32–1 outlines this group's work in three areas. The Navigation System is defined as that portion of the equipment concerned with determining the vehicle flight path. Guidance Systems relate to items which provide information for the control of the flight path, and Control Systems cover those sub-systems which actually move the space vehicle. The list in these three categories is certainly not all-inclusive, but does represent the type of work needed for today's spacecraft equipment. It is instructive to go over several of the items in some detail to illustrate the scope of the research.

NAVIGATION SYSTEMS

Figure 32–2 gives a breakdown of the research program on navigation equipment. As an example of this work, one group is devoting considerable effort to investigating the uses of TV in space and has procured a special low-light-level, image-orthicon, closed-circuit TV system for research on space vehicle position determination. One of the objectives of this program is to obtain precise angular bearings of a space vehicle in a real-time digital form suitable for direct introduction into a digital computer for determining flight path. The

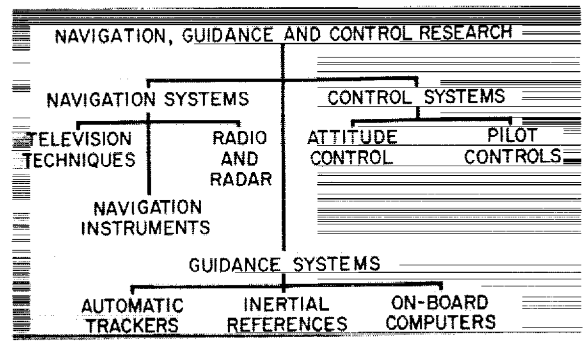

FIGURE 32–1.—Examples of equipment research.

bearings will be obtained by viewing the vehicle against a background of stars with known positions. The angular bearings to the stars and the angular difference from a star to the vehicle will be introduced into the computer. The major problems are detecting the vehicle optically at long range and obtaining the precise difference of bearings between the stars and the vehicle.

An interesting side issue being investigated is the compensation for the image motion which is due to atmospheric turbulence. Since TV images are in an electrical form they are subject to various types of filtering. There is some possibility that optimal filtering techniques can improve the resolution of astronomical pictures taken from Earth-based telescopes.

The navigation instrument studies noted on figure 32–2 are concerned with the on-board instruments used for midcourse guidance. Special sextants are being developed which will be capable of determining the angle between guide stars and the Moon or Earth or other planets to a few seconds of arc.

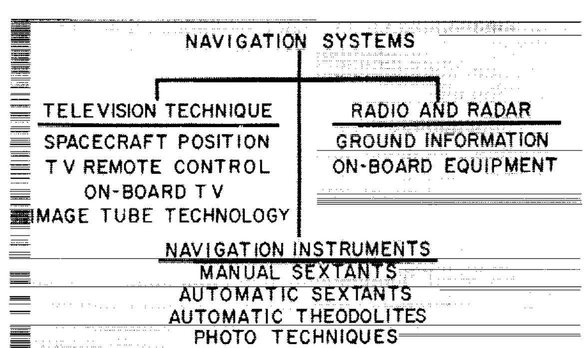

FIGURE 32–2.—Navigation equipment research.

Radio and radar systems also are important because they can be used to contribute position and velocity information to either ground-based or on-board navigation systems.

Figure 32–3 shows the research television systems used in the vehicle tracking studies. The TV camera is mounted at the focal plane of a 12-inch cassegrainian reflector telescope.

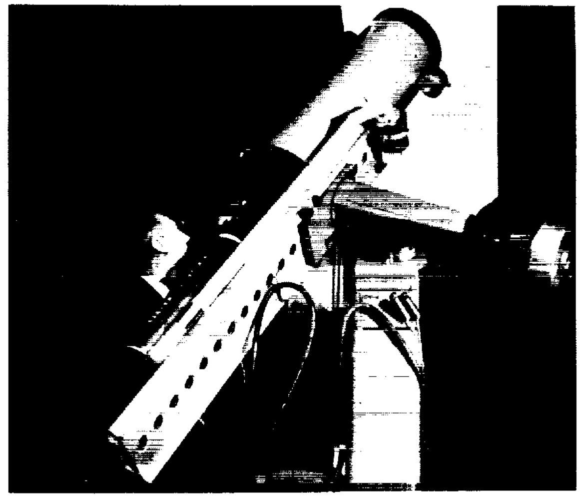

FIGURE 32–3.—Television and telescope research installation.

GUIDANCE SYSTEMS

Figure 32–4 lists the studies related to vehicle guidance equipment. An example of the work in progress in this area is the effort directed toward improving star trackers. Most star trackers used in the past have utilized mechanical methods for scanning the star image to generate the error signals needed to drive the self-tracking telescope so as to remain centered on the target star. Two scan methods are shown on figure 32–5. The mechanical scan method has limited lifetime, requires considerable power, and has other undesirable characteristics. Ames has been studying the use of the image-dissector type detector as a suitable replacement for the mechanical scan. The image dissector is a photomultiplier tube with a front electronic imaging and deflection section suitable for electronic scanning. A particularly interesting scheme uses special reticles at the tube aperture.

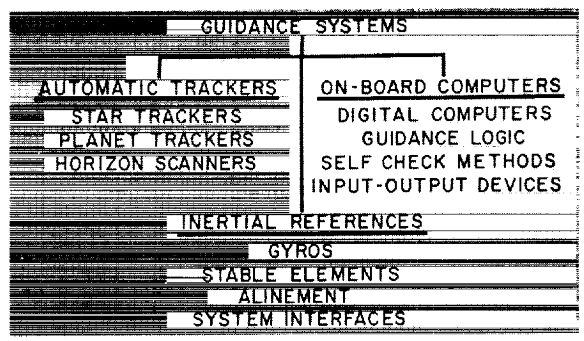

FIGURE 32–4.—Guidance equipment research.

Inertial references indicated on figure 32–4 can be used for guidance during several portions of a space mission. Present studies are concerned with both improving the equipment performance and investigating the effects of equipment errors on vehicle performance during a lunar mission.

Digital computers are an essential part of the on-board equipment for long-range missions. The many calculations needed for midcourse guidance can be accomplished only with a digital computer. Typical guidance computer studies are related to the problems of integrating the computer into the guidance system, developing guidance equations, and establishing self-checking computer procedures.

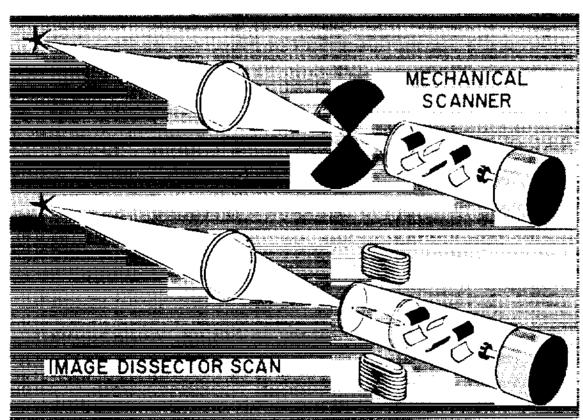

FIGURE 32–5.—Star tracker scan methods.

CONTROL SYSTEMS

Figure 32–6 itemizes certain control system equipment studies at Ames. An example of these studies is given by a current investigation of a twin-gyro attitude-control system. This control system utilizes two gyros mounted on the same frame but their rotors spin in opposite

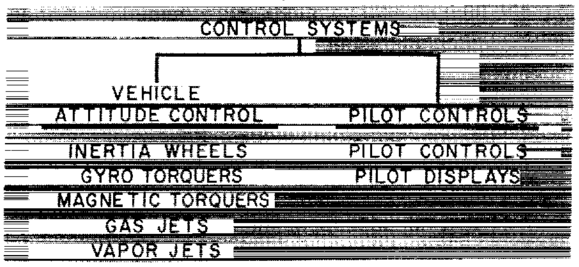

FIGURE 32–6.—Control equipment research.

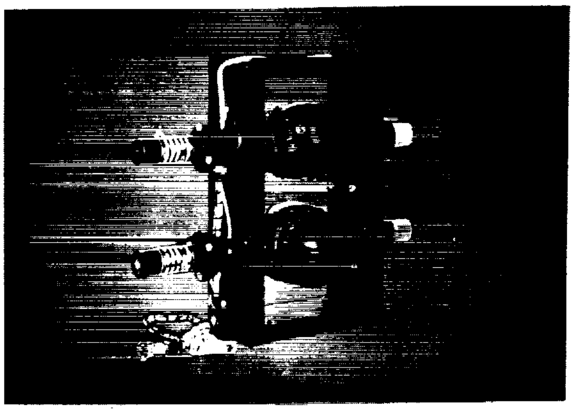

FIGURE 32–7.—Twin gyro controller.

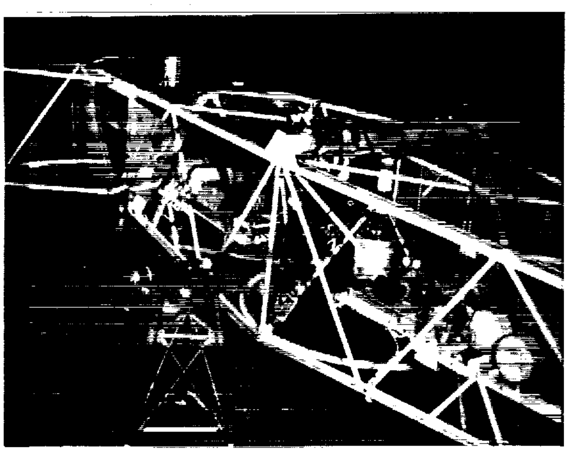

FIGURE 32–8.—Twin gyro controller experimental apparatus.

directions. These gyros are illustrated on figure 32–7. For torquing a space vehicle, the gyro gimbals are commanded to turn in opposite directions. The resulting precessional torques from the two gyros combine to react on the vehicle about the desired axis but cancel each other's cross-coupling torque about the other axis.

Figure 32–8 shows the experimental equipment associated with this study. The vehicle inertia and the torque-free environment of space are simulated by the large structure mounted on a low-friction air bearing. The pilot's seat is at one end of the structure. Three sets of special twin-gyro controllers are mounted on the frame in such a manner as to torque the frame about each of three orthogonal axes. Error signals for control are provided by a set of star trackers. The resulting stabilization system has demonstrated high dynamic performance as well as stabilizing the structure to better than ¼ second of arc.

Inertia wheels, magnetic torquers, gas and vapor jets have been studied in addition to the gyro control system described. Each has advantages and disadvantages which affect their compatibility with any particular mission. One of the objects of the studies is to define more clearly the component characteristics as an aid in choosing the optimum control method.

The pilot control studies indicated on figure 32–6 are concerned with improving the display of guidance and control information to the astronaut and optimizing his method of controlling the vehicle. A particularly interesting device under investigation is a solid-state electroluminescent panel suitable for displaying the digital computer output to the astronaut. This display consists of thin electroluminescent wafers arranged in a vertical stack. The computer output can be coded so as to illuminate the wafer whose vertical position represents the magnitude of a number. Furthermore, all the wafers below can be made to illuminate automatically so that the display is similar to that of a thermometer. The advantages are almost instantaneous time response and no moving parts.

NECESSITY OF WORKING WITH REAL HARDWARE

The previous discussion has outlined portions of a research center program on guidance and control equipment. It would be appropriate at this point to mention some precautions which must be taken in the application of this type of work. For example, care must be exercised to assure that the validity of conclusions or concepts arrived at in analytical studies are proved by sufficient tests with hardware. An example of the need for such tests was demonstrated during an Ames investigation of an alternate control system proposed for an Earth-pointing weather satellite. The stabilization system of this satellite was required to maintain one axis pointing toward the center of the Earth and another orthogonal axis pointing along the orbital path in order to scan the Earth's surface properly. Figure 32–9 is an artist's sketch of this satellite. Suitable error signals to control the axis to be pointed at the Earth's center (pitch and roll) can be obtained with a horizon scanner type of instrument, as illustrated by the figure. Unfortunately, there is no simple and direct method for generating the error signals needed to control the heading along the orbital path (yaw control). One indirect method is "gyro compassing" which uses a rate gyro mounted with its sensitive, or input axis, alined with the vehicle roll axis so as to detect a component of orbital rate when the vehicle yaws away from the correct heading. This use of a gyro, unfortunately, also results in an undesirable signal whenever the vehicle rolls. An analytical study indicated that the undesirable roll signal, in principle, could be canceled by proper circuits since roll information is available from the horizon scanner. A diagram of the proposed system is shown on figure 32–10. As illustrated, the gyro generates a signal composed of a component equal to the product of orbital rate and yaw angle, $\omega_0\psi$, and a component equal to the roll rate $\dot{\varphi}$. Only the yaw signal is desired. The horizon scanner generates roll position information which can be converted approximately into roll rate by the network shown. If appropriate gains are used, the roll-rate signal generated by the horizon scanner can be subtracted from other gyro signal leaving only the desired yaw-angle component. Such a system was studied on an analog computer which indicated the system performance to be satisfactory. Nevertheless, in view of other experience with cancellation techniques, it was decided to further check the method by using the same analog computer to simulate most of the system but to use a real horizon scanner for providing the

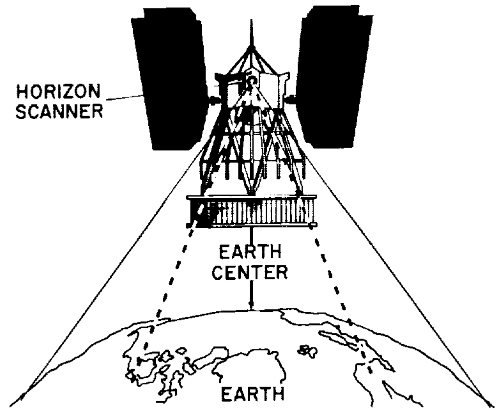

FIGURE 32–9.—Horizon scanner control.

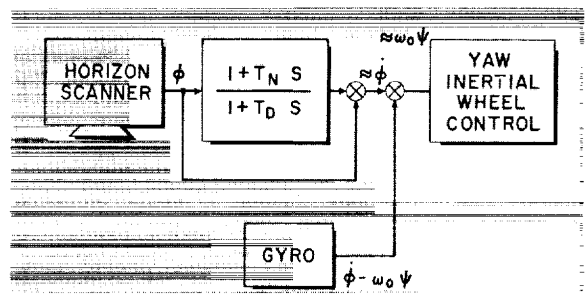

FIGURE 32–10.—Block diagram of yaw control scheme.

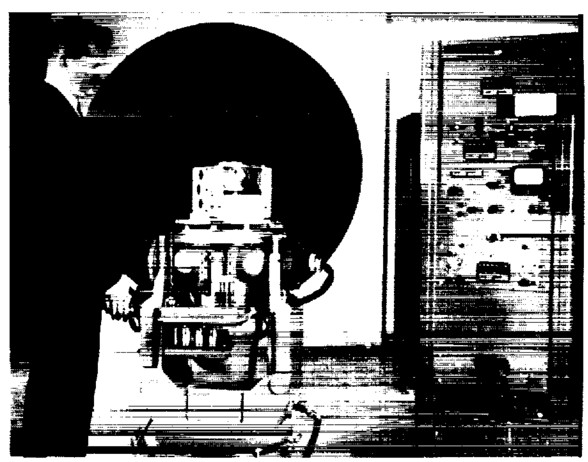

FIGURE 32–11.—Rate table and horizon scanner apparatus.

error signals. This was accomplished by driving a precision rate table from the analog computer and mounting the horizon scanner on the rate table to "look" at a circular hot plate simulating the Earth as shown in figure 32–11. The results were now considerably different, since the real horizon scanner had sufficient noise in its output to make the cancellation technique unworkable.

DIFFICULTIES ASSOCIATED WITH REAL HARDWARE

In actual space-vehicle project work it is difficult to investigate radically new concepts for guidance and control equipment. Although the vehicle mission may involve extremely advanced space experiments or explorations, the vehicles themselves must be as reliable as possible. Consequently, when component development is tied to a particular project, several constraints are immediately imposed. These are outlined below:

1. Project time schedule.
2. "Tried and true" concept.
3. Major system design restrictions.
4. Limited freedom for subsystem designers.

The listed constraints are the consequence of the necessity of developing reliable vehicles and associated systems for navigation, guidance, and control in as short a time as practicable. The people responsible for these programs would like to use the latest or most promising new techniques but the harsh realities do not allow this luxury.

Considering then the restraints in order, the first limits the time allowable for development and, consequently, severely restricts any investigation of promising but speculative new techniques. In a similar fashion, the second item states that, insofar as possible, only proven techniques be used. The reason underlying this constraint is that space vehicles have minimum redundancy so if any of the many one of a kind components fail, the entire mission is lost. Consequently, both the NASA and the contractors must insist on proven techniques whenever possible. The third constraint results from the need to choose an over-all system to meet the operational requirements of the vehicle. In the process, the performance and configuration of subsystem components, such as star trackers, may be specified within rather narrow limits. The final constraint reflects the subcontractor's dilemma. In spite of his desire to develop the ultimate in component performance, he is faced with two factors: a short time schedule and a detailed knowledge of how he built his last component. The tendency, therefore, is to update old designs to meet new requirements.

These project constraints emphasize the necessity of conducting advanced research away from the pressures of project development. Within the NASA the research centers provide this function and it is for this reason that the Centers concentrate on advanced techniques rather than hardware development. Furthermore, this project development situation has a bearing on the universities' role in the space research effort. Since the universities are not under such constraints, a splendid opportunity is provided for them to contribute radically new concepts to the guidance and control equipment area. For example, decisions to investigate new and exotic but speculative concepts do not have to be weighed against the consequence of failure. As a result, an atmosphere conducive to the exploration of new ideas can prevail.

SUMMARY

Typical research programs on guidance and control equipment have been discussed. Some of the hardware difficulties and project constraints have been outlined. It appears that the universities are in an excellent position because of their wide choice of research talent and lack of constraints inherent in funded space programs to make major contributions toward advancing the state of the art.